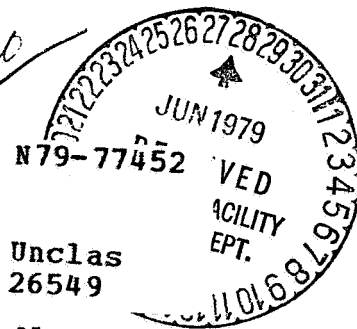


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MISSION A-004 SPACECRAFT 002 (North American  
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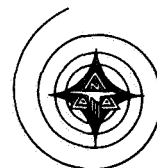
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APOLLO MISSION A-004  
SPACECRAFT 002

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NAS9-150

29 January 1965

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## TECHNICAL REPORT INDEX/ABSTRACT

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## ABSTRACT

This document defines the engineering test requirements and test plans for Apollo Power-On Tumbling Boundary Abort Mission A-004. It will be the basic document for NASA preparation of the mission directive for this test.

065-4530

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## FOREWORD

This document describes the requirements and plans for the preparation and conduction of the Apollo power-on tumbling boundary planned abort mission with Spacecraft 002. The mission is scheduled to be performed at the White Sands Missile Range as the sixth and final test in a series of flight tests of the Apollo vehicle launch escape configuration. The vehicle configuration at launch will consist of a spacecraft launch escape subsystem with canards and boost protective cover, spacecraft command and service modules, a special adapter for service module to booster transition, and a Little Joe II boost vehicle.

Major contributions to this document were furnished by System Dynamics, Development Analysis, Structures and Materials Development Analysis, Systems Integration, and Apollo Test and Operations.

Comment concerning this document should be directed to the Test Requirements Section of Apollo Development Analysis Engineering, NAA, Downey, extension 3622 or 3623.





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## 1.0 INTRODUCTION

### 1.1 PURPOSE

The "Vehicle Test Plan, Apollo Mission A-004" presents the engineering test requirements and objectives, the plans for vehicle checkout prior to the flight, the tasks to be accomplished during the mission, and data requirements and reporting procedures. This information will be used by NASA to prepare the mission directive.

### 1.2 PRECEDENCE OF DOCUMENTS

This test plan takes precedence over all other contractor reports and documents concerned with Spacecraft 002 testing; and, in turn, it is superseded by the NASA Project Apollo flight mission directive for Mission A-004.

### 1.3 DEFINITIONS

Terms used in this document are defined in Appendix A.



## 2.0 FLIGHT TEST OBJECTIVES

### 2.1 MISSION SUMMARY

Apollo Mission A-004 will be a planned abort mission in the power-on tumbling boundary region. This region is defined by abort initiation conditions in which the combined aerodynamic and launch escape motor exhaust pressures of tumbling abort imposed on the command module surface affect a pressure differential across the command module outer structure (outer surface to cavity between inner and outer structures) that approaches the design limit range of 9.0 to 11.1 psi. The mission is assigned for performance with Spacecraft 002 at WSMR; Spacecraft 010 is the backup vehicle for the mission. The launch escape vehicle will be launched by a Little Joe II booster to approximately 75,000 feet, at which point the abort will be initiated. A pitch-up will be performed by the booster just before abort initiation to ensure escape vehicle tumbling during the abort. An apogee of approximately 120,000 feet will be reached. Tumbling will continue up through the apogee and down through the upper region of descent until orientation and stabilization are accomplished by the canards. Deceleration for vehicle touchdown and recovery 42 miles from the point of launch will be accomplished by the earth landing subsystem.

### 2.2 TEST OBJECTIVES

The test objectives contained in this section constitute the achievements to be realized through performance of Mission A-004. The test objectives for all missions of the Apollo abort flight program at WSMR (abort through launch escape subsystem performance) are presented on Figure C-1 of Appendix C.

First-order test objectives are as follows:

1. Demonstrate satisfactory launch escape vehicle (LEV) performance for an abort in the power-on tumbling boundary region
2. Demonstrate the structural integrity of the LEV airframe structure for an abort in the power-on tumbling boundary region

Second-order test objectives are as follows:

1. Demonstrate the capability of the canard subsystem to satisfactorily reorient and stabilize the LEV heat shield forward after a power-on tumbling abort



2. Demonstrate the structural capability of the production boost protective cover (BPC) to withstand the launch environment
3. Determine the static loads on the command module during the launch and the abort sequence
4. Determine the dynamic loading of the command module inner structure
5. Determine the dynamic loads and the structural response of the service module during launch
6. Demonstrate the capability of the command module forward heat shield thrusters to separate the forward heat shield satisfactorily after the tower has been jettisoned by the tower jettison motor
7. Determine the static pressures imposed on the command module by free-stream conditions and launch escape subsystem (LES) motor plumes during a power-on tumbling abort

Third-order test objectives are as follows:

1. Demonstrate satisfactory separation of the launch escape vehicle from the service module
2. Demonstrate the satisfactory operation and performance of the earth landing subsystem (ELS) with a spacecraft vehicle
3. Obtain data on the structural response of the command module during earth landing subsystem sequence
4. Obtain thermal data on the boost protective cover during a power-on tumbling abort
5. Obtain acoustical noise data inside the command module at an astronaut station



### 3.0 TEST OBJECTIVE EVALUATION CRITERIA

This section restates verbatim the test objectives of Section 2.0, describes the issues associated with each objective, and specifies the data procurement instrumentation and other documentation media for recording the accomplishment of objectives.

Diagrams of test vehicle instrumentation points are included in this section to afford greater comprehension of instrumentation locations. These diagrams, Figures 3-1 through 3-10, are copies of illustrations in SID 63-502, Apollo Measurement Requirements, Spacecraft 002.

#### 3.1 FIRST-ORDER TEST OBJECTIVE CRITERIA

1. Objective: Demonstrate satisfactory launch escape vehicle (LEV) performance for an abort in the power-on tumbling boundary region.

Criteria: Upon command to abort, the launch escape subsystem (LES) will be required to propel the spacecraft configuration launch escape vehicle on an escape course to a safe distance beyond the service module - Little Joe II booster stack for descent and eventual recovery. Since LEV tumbling is expected to commence at abort initiation because of instability created by a change in LEV angle-of-attack (Little Joe II pitchup), by a shift in the center of aerodynamic pressure on the LEV by escape motor exhaust plume envelopment, and by short-duration thrust of the pitch control motor, the escape motor will propel the LEV from the booster in a curving trajectory. If the rate of tumbling is rapid or if motor combustion duration is appreciable, part of escape motor thrust might be expended in driving the LEV back toward the Little Joe II booster. However, preliminary analysis indicates that the maximum tumbling rate expected of the first tumble should be no greater than 90 degrees in three seconds. Since effective combustion of the launch escape motor will be expended in less than four seconds, as indicated on Figure 4-6, it is concluded that the LEV will be driven only on a course divergent from the booster. The order of distance between the LEV and the booster required to constitute satisfactory abort performance of the LEV and the order of time duration required to achieve the necessary separation are undefined.

Preliminary analysis also indicates that the gravity forces induced during power-on tumbling should be well within astronaut limitations.



Attitude gyro, rate gyro, accelerometer, and escape motor chamber pressure instrumentation data plus ground stationed cine camera and cine theodolite recordings will be obtained to define the LEV trajectory. The superimposition of the LEV trajectory on the Saturn trajectory will be analyzed for successful performance accomplishment. Data of the same parameters and camera and theodolite recordings minus that of motor chamber pressure will be used for analysis of gravity force characteristics and comparison with astronaut limitations. Locations of LEV accelerometer sensors are indicated on Figures 3-1 and 3-2.

2. Objective: Demonstrate the structural integrity of the LEV airframe structure for an abort in the power-on tumbling boundary region.

Criteria: The LEV is expected to encounter severe aerodynamic, inertial, and escape motor exhaust blast forces during power-on tumbling. The exhaust blast forces on the conical surface of the command module can be especially great, since tumbling will tend to drive one side of the conical surface into the plume pattern in a normal or near normal direction. The exhaust cone will be augmented by free stream pressure external of the cone against the direction of tumble. The ability of the vehicle structure to remain intact in the tumbling environment will be analyzed from photographic recordings of the on-board camera shown in Figure 12-2 and of ground stationed cameras.

### 3.2 SECOND-ORDER TEST OBJECTIVE CRITERIA

1. Objective: Demonstrate the capability of the canard subsystem to satisfactorily reorient and stabilize the LEV heat shield forward after a power-on tumbling abort.

Criteria: The canards will be deployed 11 seconds after abort initiation following escape motor burnout. However, end-over-end LEV tumbling is expected to continue during coastup through the apogee and on down in the descent until the canards become sufficiently aerodynamically effective to arrest tumbling and orient the LEV with the aft heat shield toward earth. This is expected to occur in the altitude range from 90,000 to 75,000 feet. The LEV will then probably oscillate about its trim angle-of-attack in near 180-degree sweeps and diminish gradually with decreasing altitude until satisfactory stabilization is achieved from 50,000 to 25,000 feet altitude. Stabilization and proper orientation are necessary to enable proper deployment of the earth landing subsystem (ELS) parachutes.

The canards are mechanized for deployment at 11 seconds after initiation of any type of abort in which the LES is employed. This value was selected to accommodate the time-critical pad abort sequence.

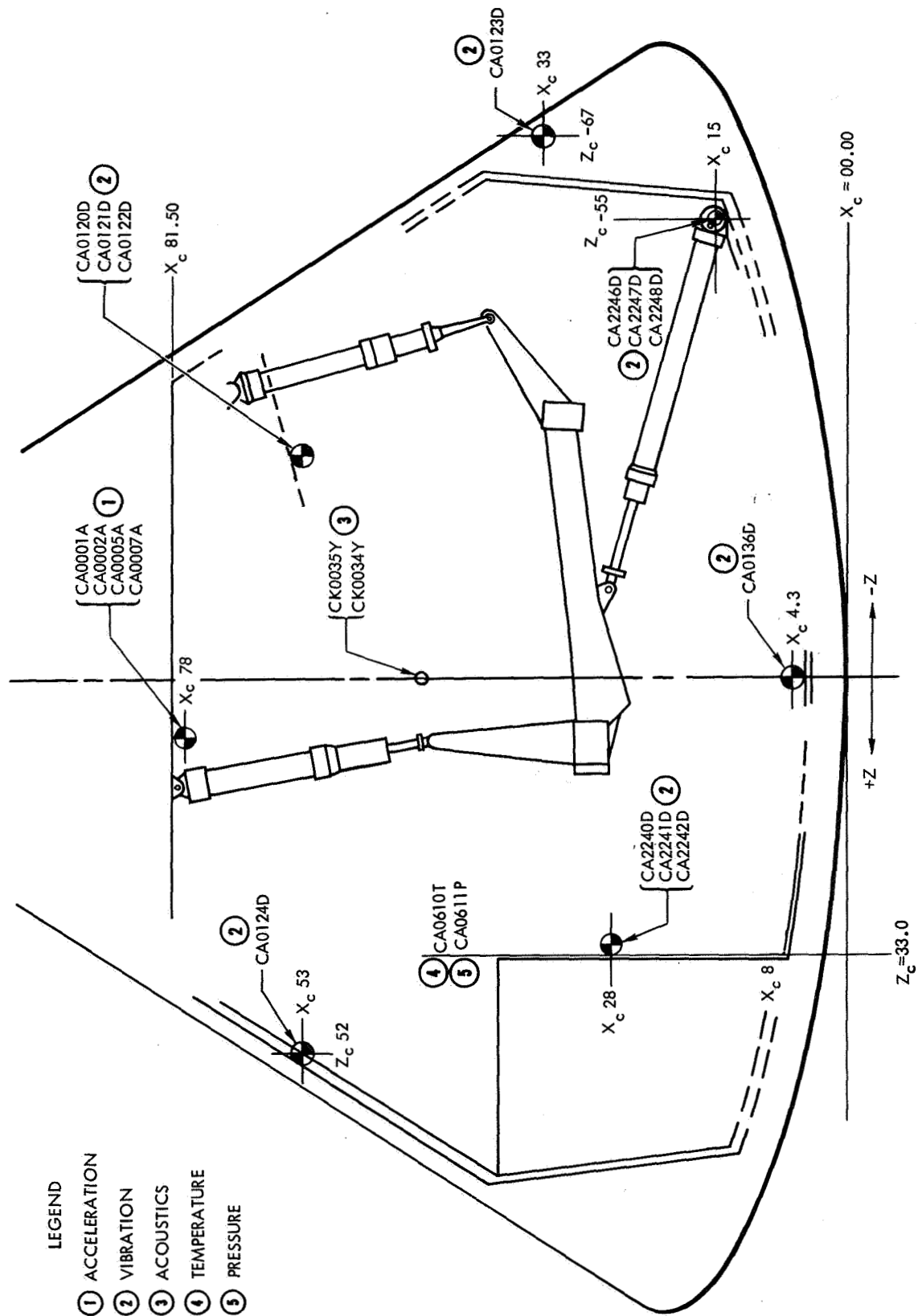


Figure 3-1. CM Internal Measurement Locations

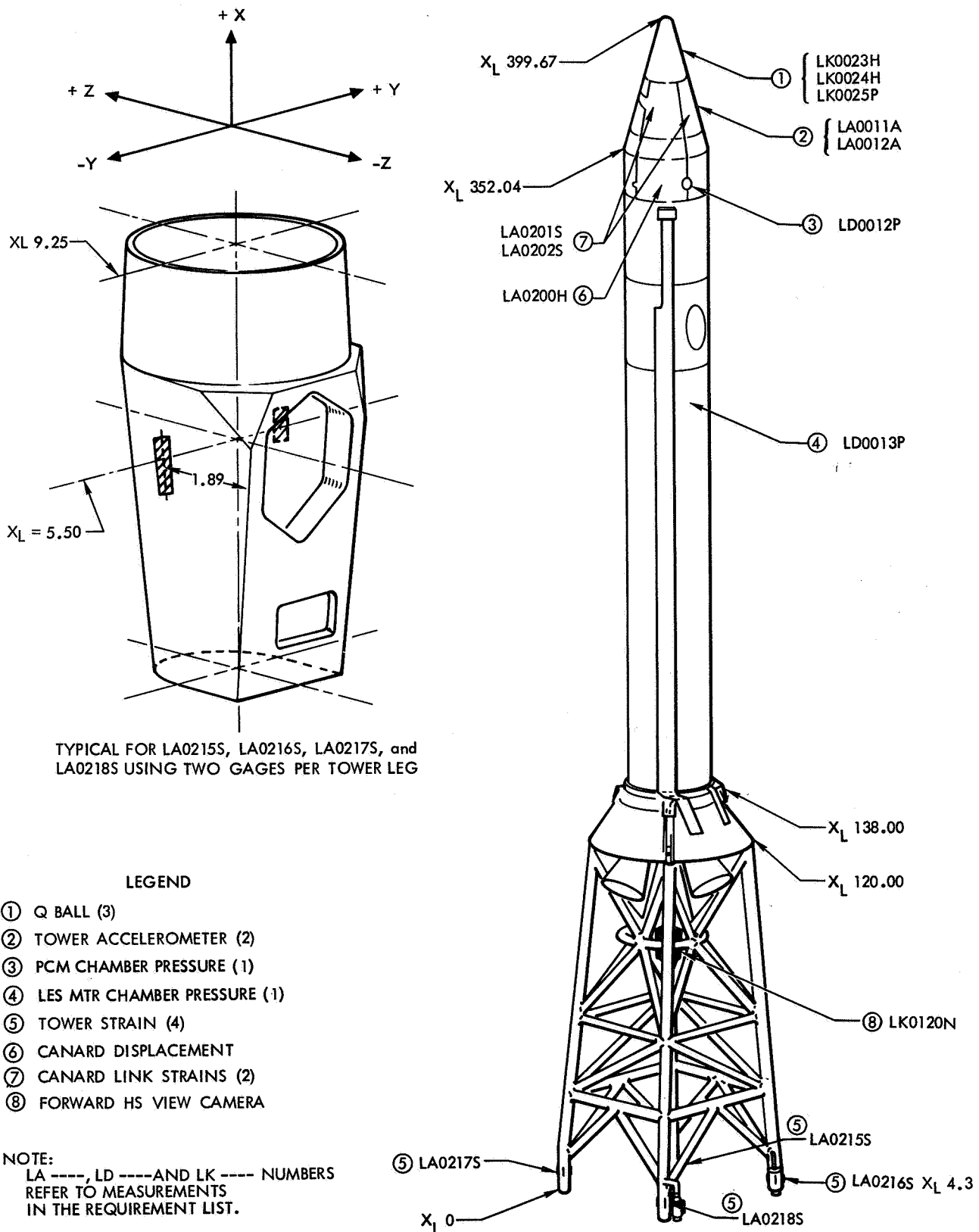


Figure 3-2 . Launch Escape Subsystem Measurement Locations



Canard performance will be analyzed from attitude gyro, rate gyro, and accelerometer instrumentation data plus ground stationed camera recordings.

2. Objective: Demonstrate the structural capability of the production boost protective cover (BPC) to withstand the launch environment.

Criteria: The design purpose of the BPC is to shield the command module conical surface from aerodynamic heating during the launch phase. Although the launch of this mission will constitute only part of the normal mission launch phase, this will be the first time that a spacecraft configuration BPC will be exposed to actual launch environment. It will be necessary to demonstrate that throughout this launch the spacecraft configuration BPC remained intact with no penetrations and remained in position on the command module. Performance of the BPC will be analyzed from on-board tower-installed and ground-stationed camera recordings.

3. Objective: Determine the static loads on the command module during the launch and the abort sequence.

Criteria: Static loads are structural-type loads of very low frequency variation. The command module is expected to encounter severe aerodynamic, inertial, and escape motor exhaust blast forces during power-on tumbling resulting in high static loading of command module structural members. Since this will be the first spacecraft configuration command module to be launched in flight, the load information obtained will be a basis of subsequent command module design. Command module static loads during launch and abort will be determined from strain instrumentation data. The strain instrumentation locations are indicated on Figures 3-3 through 3-6.

4. Objective: Determine the dynamic loading of the command module inner structure.

Criteria: The command module will be subjected to aerodynamic noise, acoustic noise, and mechanically transmitted vibration throughout the flight, each contributing to the overall dynamic load on the structure. Aerodynamic noise is expected to be of greatest intensity through the transonic region of the launch phase. Acoustic noise probably will be highest during LES motor burning. Dynamic loading and response characteristics of the spacecraft configuration command module structure will be determined for the first time from vibration, accelerometer, and strain instrumentation on inner structural members. Some of the instrumented points are indicated on Figures 3-1 and 3-4 through 3-6.







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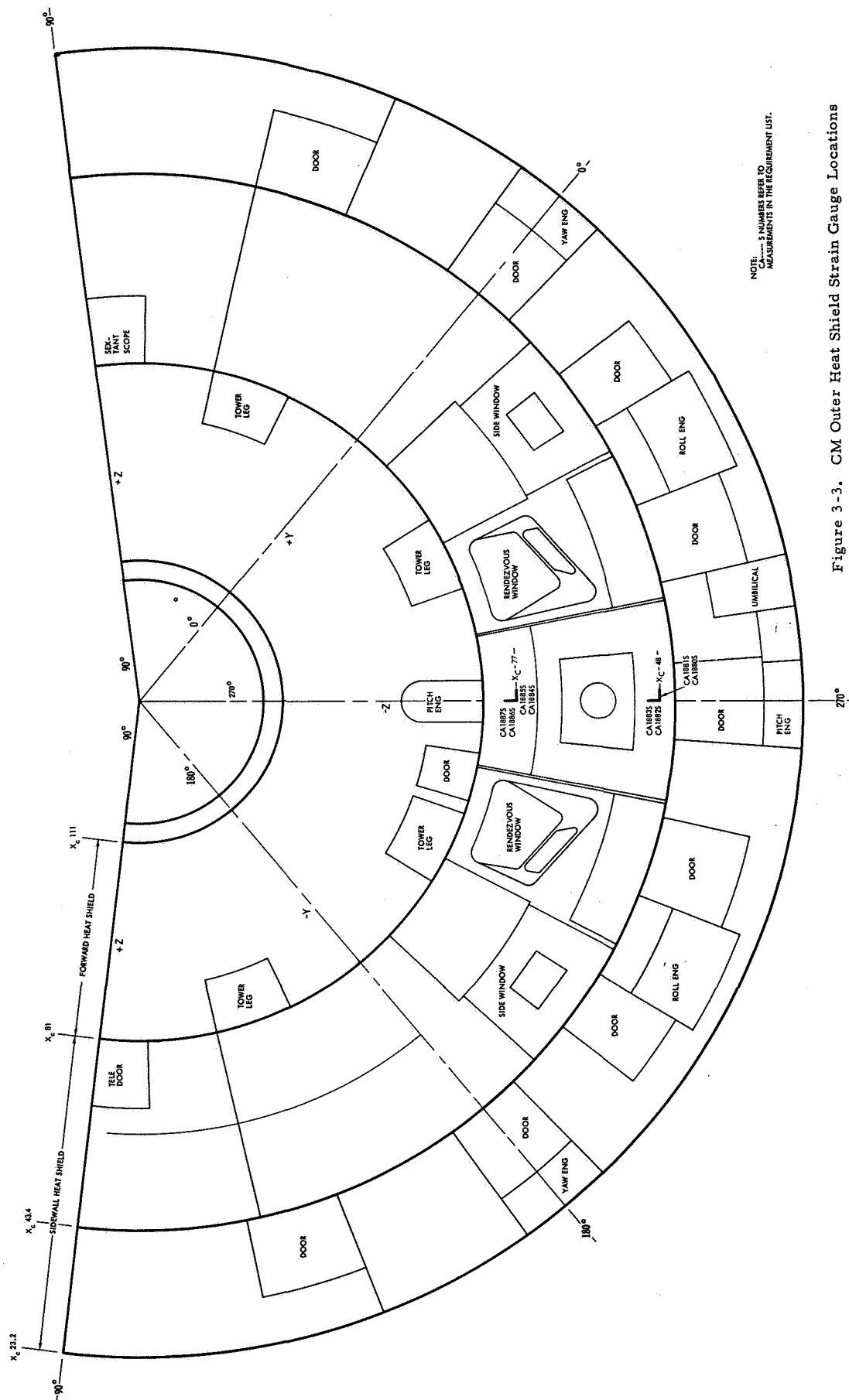


Figure 3-3. CM Outer Heat Shield Strain Gauge Locations

3-7, 3-8  
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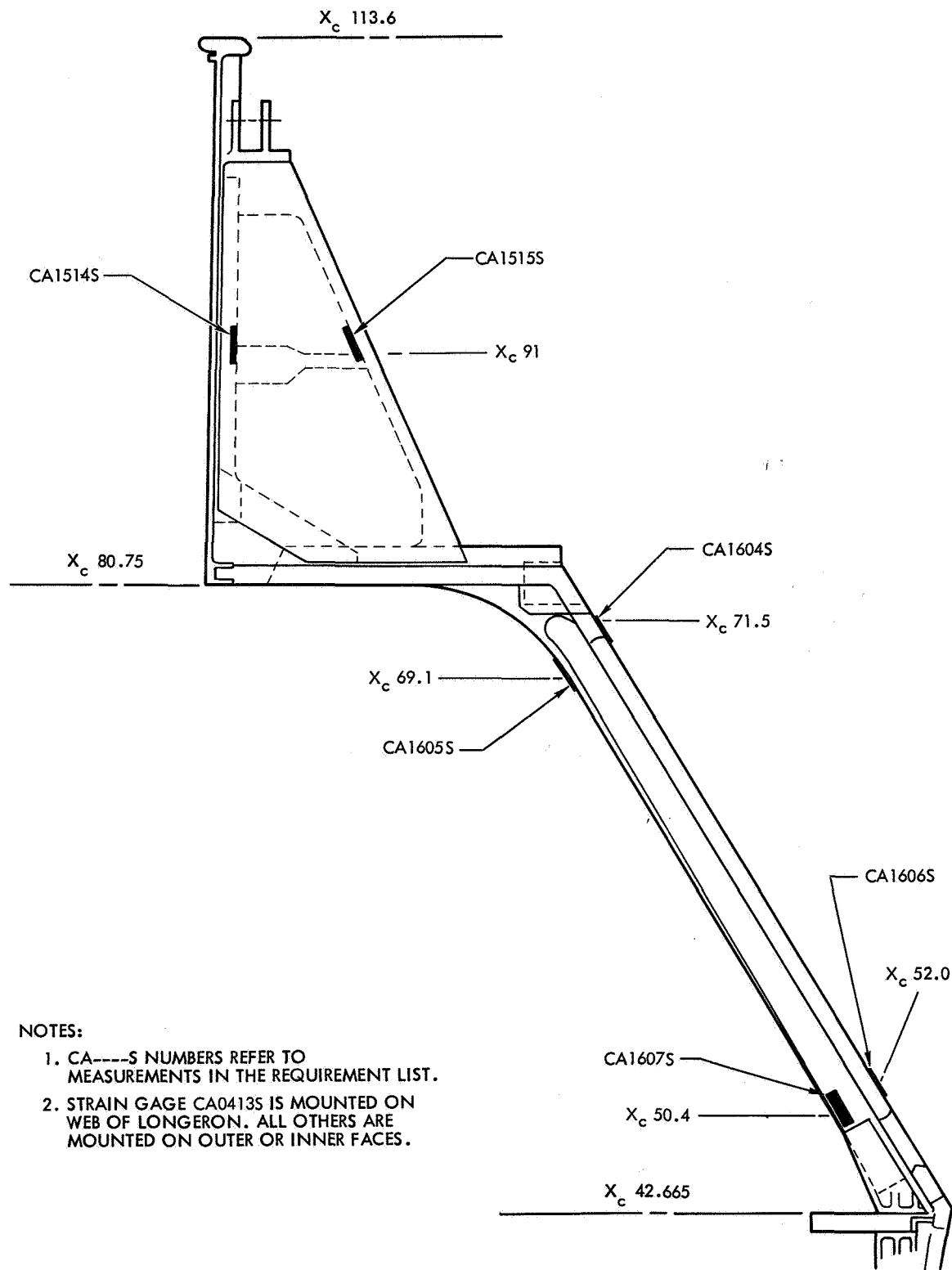


Figure 3-4. CM Inner Structure—Forward Sidewall  
Longeron 4 Strain Gage Locations



## NOTES:

1. ALL GAGES READ PARALLEL TO LONGERON AXIS.
2. CA----S NUMBERS REFER TO MEASUREMENTS IN THE REQUIREMENT LIST.
3. NUMBERS IN CIRCLES INDICATE LONGERON NUMBER GAGE IS MOUNTED ON.

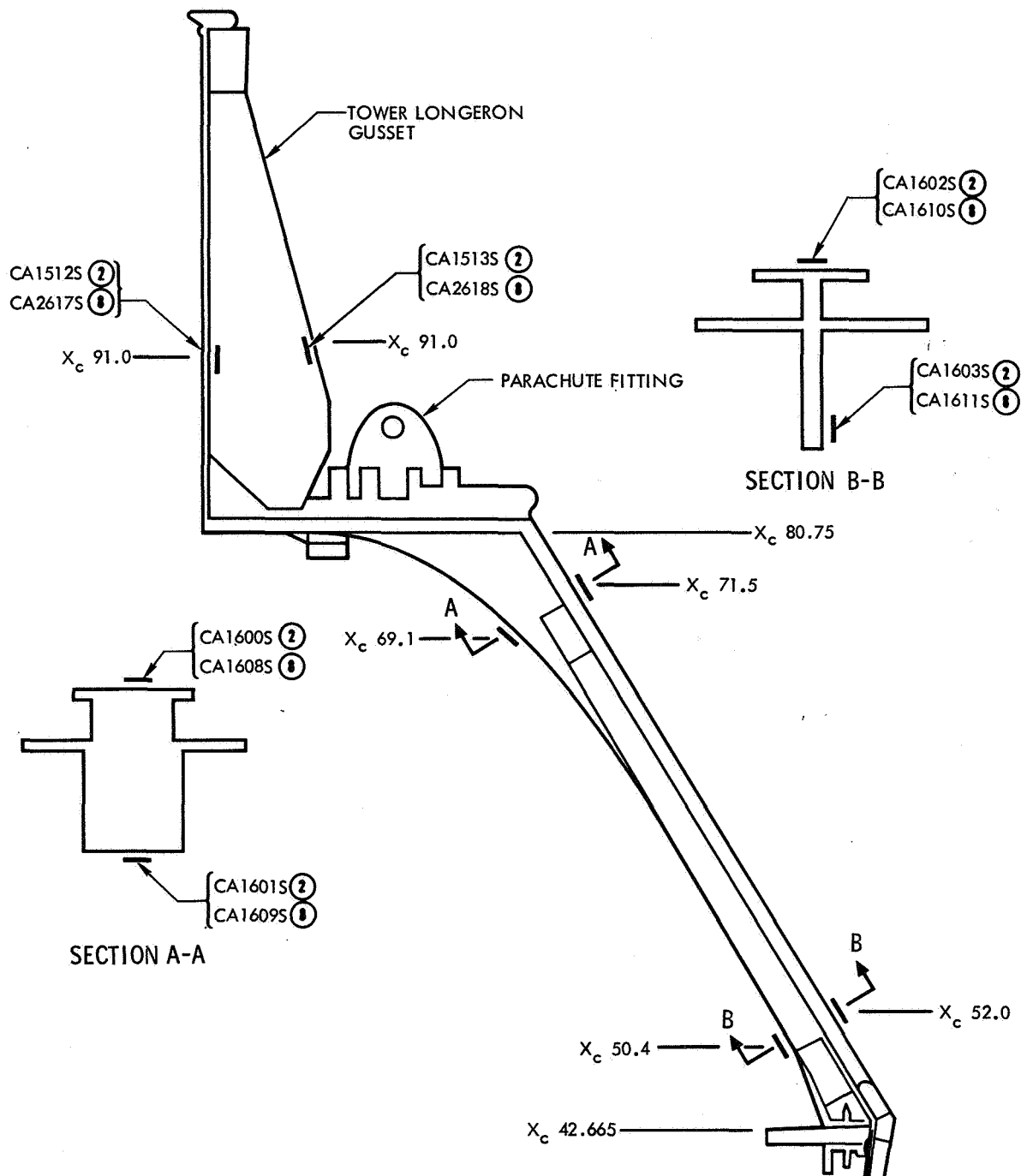


Figure 3-5. Strain Gage Locations on Longerons 2 and 8

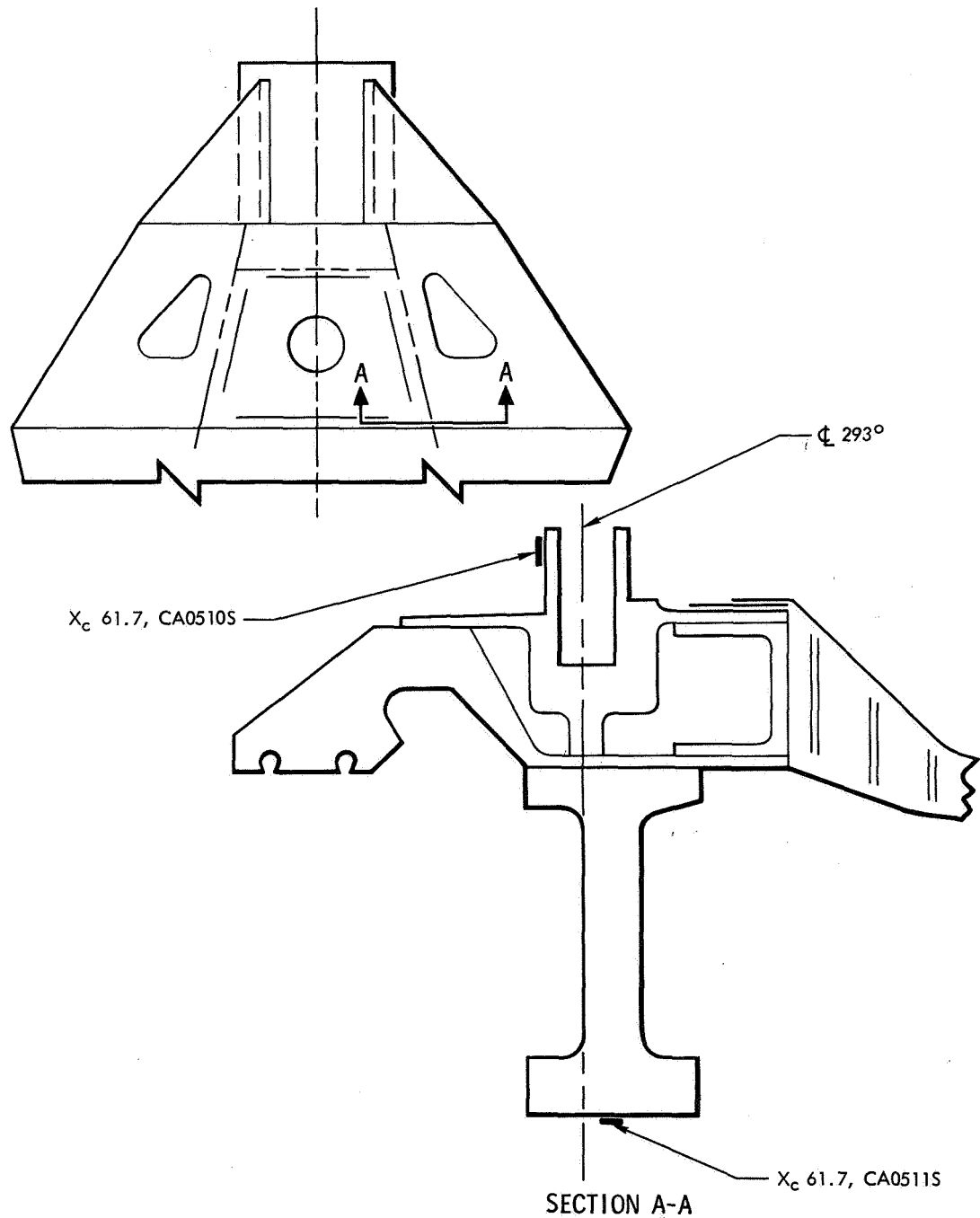


Figure 3-6. Strain Gage Locations on Right-Hand Beam of Main Hatch



5. Objective: Determine the dynamic loads and the structural response of the service module during launch.

Criteria: Aerodynamic noise and mechanical vibration will be encountered by the service module during launch. The highest levels of aerodynamic noise will probably be encountered through the transonic region. The RCS engine quads, by their protuberant installations, will likely induce concentrations of aerodynamic noise and vibration on localized surface areas of the service module which will affect the overall dynamic load. Dynamic loading and response characteristics of the spacecraft configuration service module structure will be determined for the first time from vibration accelerometer and strain instrumentation on the service module. Some of the instrumentation points are indicated on Figures 3-7 and 3-8.

6. Objective: Demonstrate the capability of the command module forward heat shield thrusters to separate the forward heat shield satisfactorily after the tower has been jettisoned by the tower jettison motor.

Criteria: The mission sequencer subsystem is mechanized to transmit the thruster firing signal 0.4 seconds after transmission of the LES jettison motor firing signal. This delay of forward compartment exposure is intended to protect the earth landing equipment from the boost protective cover as it sweeps and brushes by the forward compartment upon extraction by the tower during LES jettison. Forward heat shield ejection by the thrusters is required immediately after LES jettison in order to allow ample time for command module deceleration by parachute series deployment. Demonstration of satisfactory heat shield ejection will be documented by ground photographic coverage.

7. Objective: Determine the static pressures imposed on the command module by free-stream conditions and LES motor plumes during a power-on tumbling abort.

Criteria: Power-on tumbling in the tumbling boundary region will drive one side of the command module conical surface into the launch escape motor exhaust blast in a normal or near normal impingement, resulting in high pressure impact on the surface facing the direction of tumble. Analysis indicates that these surface pressures might be sufficient to create a pressure differential between this surface and the space cavity between inner structure and outer shell (vented to free stream pressure) that approaches the design limit range of approximately 9 to 11 psi.

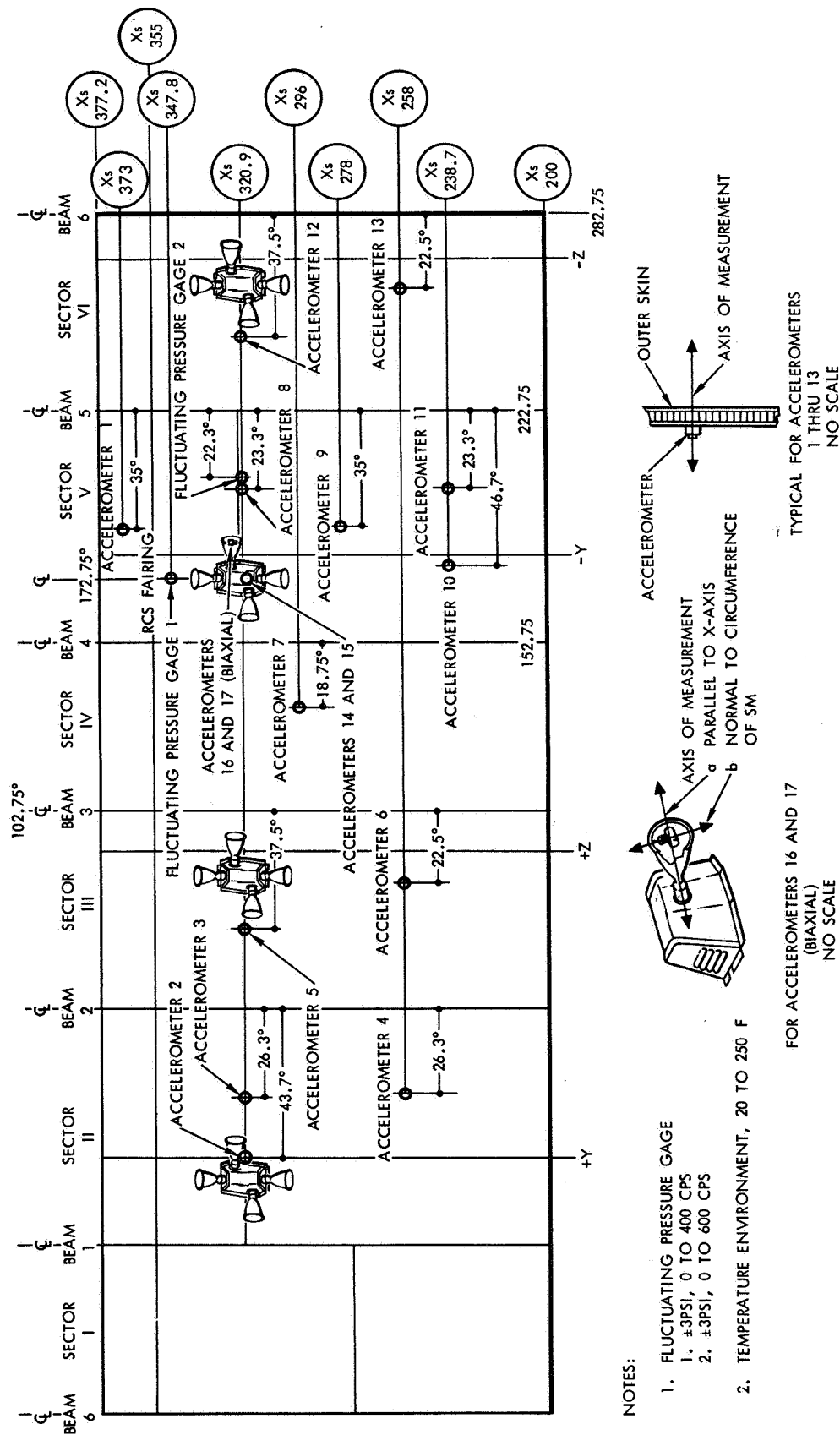
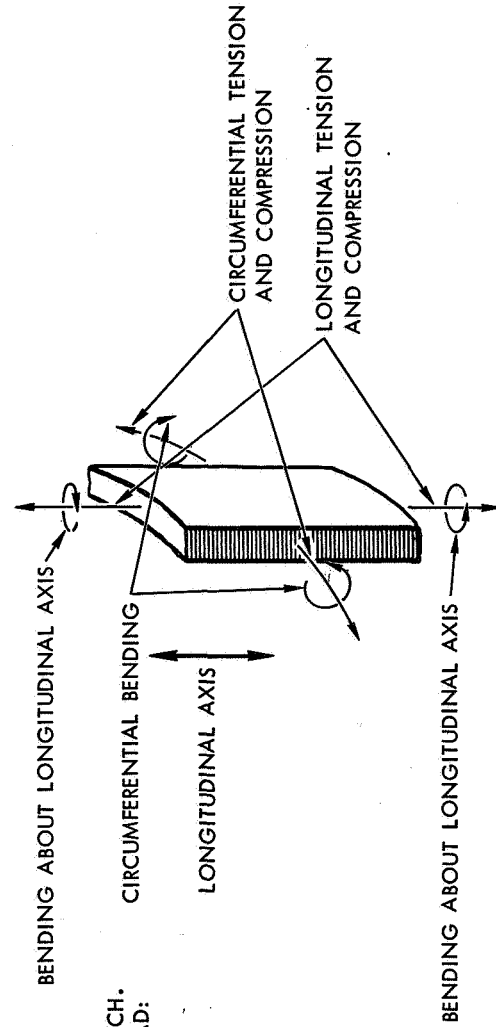
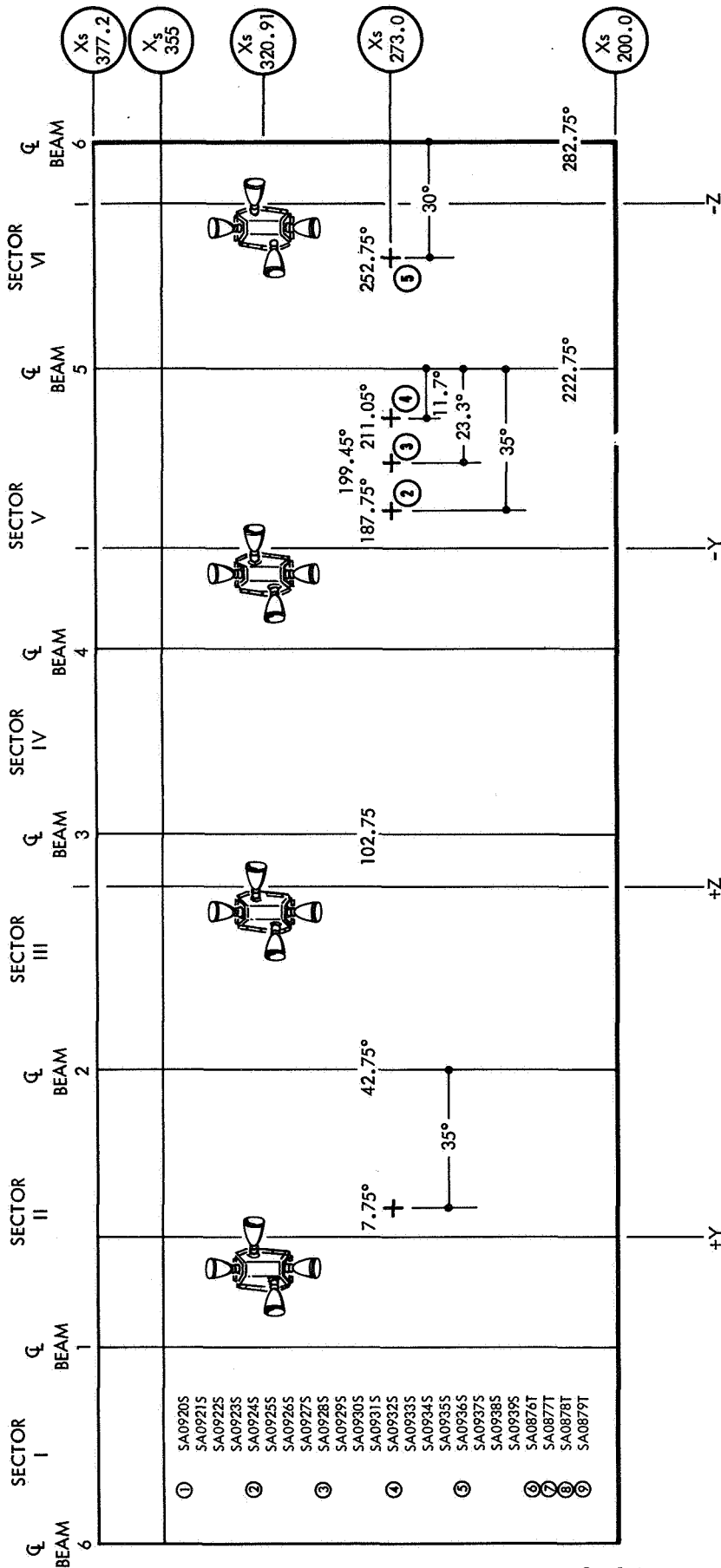


Figure 3-7. SM Accelerometer and Fluctuating Pressure Measurement Locations



## NOTES:

- ALL STRAIN GAGE RANGES ARE  $\pm 7000$  MICROINCHES PER INCH.
- STRAIN GAGES AT EACH LOCATION ARE ARRANGED TO READ:
  - BENDING STRAIN ABOUT LONGITUDINAL AXIS
  - BENDING STRAIN ABOUT CIRCUMFERENTIAL AXIS
  - TENSILE AND COMPRESSIVE STRAIN, LONGITUDINAL
  - TENSILE AND COMPRESSIVE STRAIN, CIRCUMFERENTIAL
- ALL STRAIN GAGES ARE COMPENSATED FOR TEMPERATURE EFFECTS, 20 TO 250 F.
- ALL TEMPERATURE GAGE RANGES ARE FROM +32 TO +482 F.

Figure 3-8. SM Temperature and Strain Gage Locations





Pressure values on the command module outer surface will be obtained from pressure instrumentation points indicated on Figures 3-9 and 3-10. Pressure in the space cavity between inner and outer structures will be derived from reference static pressure instrumentation.

### 3.3 THIRD-ORDER TEST OBJECTIVE CRITERIA

1. Objective: Demonstrate satisfactory separation of the launch escape vehicle from the service module.

Criteria: Spacecraft 002 will be the first Apollo flight vehicle to be equipped with the command and service module electrical umbilical and plumbing hard lines. Therefore, this will be the first flight on which the umbilical and hard lines must be properly cut in order to achieve satisfactory command module separation from the service module. Demonstration of this event will be documented for analysis by ground-stationed cameras.

2. Objective: Demonstrate the satisfactory operation and performance of the earth landing subsystem (ELS) with a spacecraft vehicle.

Criteria: The spacecraft 002 command module will be the first near-control-weight command module to be decelerated and returned to earth from actual flight by the ELS. The maximum descent design rate velocity of 33 feet per second at 5000-foot elevation (4200-foot WSMR elevation) is required to be achieved by two main parachutes functioning properly. The descent rate is expected to be approximately 26 feet per second with three main parachutes in proper function. ELS performance with the near-control-weight command module will be analyzed from attitude gyro, rate gyro, and accelerometer data plus ground-stationed camera recordings.

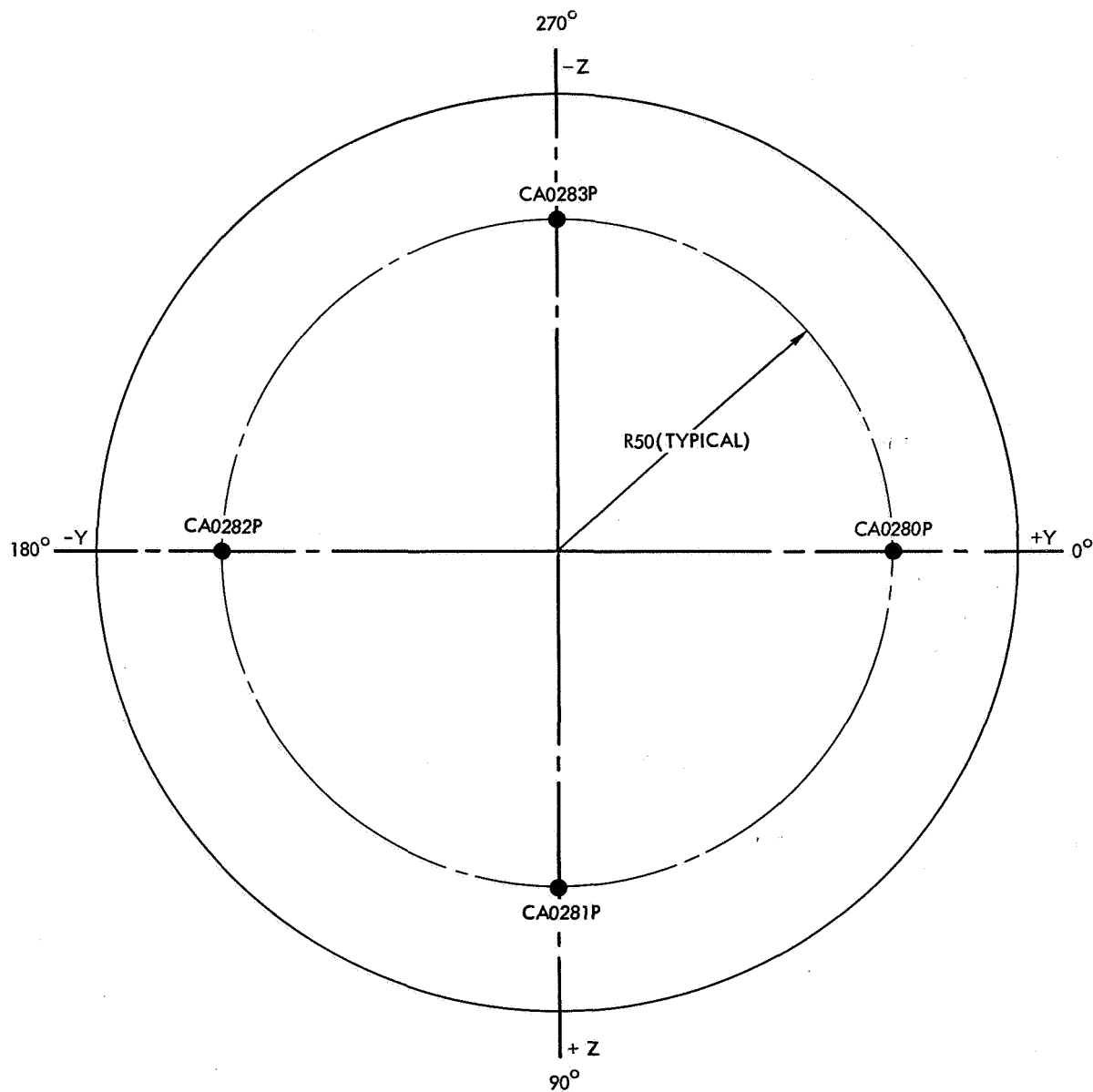
3. Objective: Obtain data on the structural response of the command module during earth landing subsystem sequence.

Criteria: Parachute deployment shock suspension loads are expected to be localized around the forward hatch area of the command module inner structure. Side loads also might be induced if parachute initial line stretch occurs while the command module is in an oscillating swing angle from the direction of descent. Structural response data on a spacecraft command module in actual flight will be obtained for the first time from vibration, accelerometer, and strain instrumentation.

4. Objective: Obtain thermal data on the boost protective cover during a power-on tumbling abort.



3-17, 3-18  
SID 64-2174



## NOTE

1. CA---P NUMBERS REFER TO MEASUREMENTS IN THE REQUIREMENT LIST.
2. VIEW IS LOOKING FORWARD.

Figure 3-10. Aft Heat Shield Base Pressure Measurement Locations



Criteria: This will be the first spacecraft configuration boost protective cover to be installed on the command module for actual flight. Thermal data of the cover will be analyzed for effectiveness of heat shielding qualities. These data will be obtained from temperature and heat flux instrumentation installed on the conical surface of the command module. Locations of the instrumented points are indicated on Figure 3-9.

5. Objective: Obtain acoustical noise data inside the command module at an astronaut station.

Criteria: Sound attenuation properties across the spacecraft configuration command module structure in flight will be investigated with data obtained from acoustical instrumentation inside the command module on the center couch platform. The command module decibel allowance in the speech interference level (300 to 4800 cycles per second) with all equipment operating in normal flight is 55 decibels.



## 4.0 TEST VEHICLE CONFIGURATION

## 4.1 VEHICLE SUBSYSTEM PRIORITIES

Test Vehicle Subsystems	Configuration	Priority
Launch escape subsystem		
Q-ball	Complete	Primary
Pitch control motor	Complete	Primary
LES jettison motor	Complete	Primary
Launch escape motor	Complete	Primary
Canard	Complete	Primary
Tower	Complete	Primary
Boost protective cover	Complete	Primary
Earth landing subsystem		
Drogue parachutes	Complete	Primary
Pilot parachutes	Complete	Primary
Main parachutes	Complete	Primary
Primary structures		
Command module	Complete	Primary
Service module	Complete	Primary
Adapter	Simulated	Primary
Retention and separation subsystem		
CM-SM separation	Complete	Primary
CM-tower separation	Complete	Primary
CM-forward heat shield separation	Complete	Primary
Sequencer subsystem		
Mission sequencer	Interim	Primary
Tower sequencer	Interim	Primary
ELS sequencer	Complete	Primary



Test Vehicle Subsystems	Configuration	Priority
Electrical power subsystem		
Batteries and power distribution	Interim	Primary
Communication and instrumentation subsystem		
Telemetry and antenna	Interim	Primary
Onboard recorder	Interim	Primary
End instrumentation and signal conditioners	Interim	Primary and secondary
C-band transponder and antenna	Interim	Primary
Camera	Interim	Primary
Launch vehicle subsystems		
Propulsion	Interim	Primary
Stabilization and control	Interim	Primary
Abort timer	Interim	Primary

See Appendix A for definitions of terminology.

#### 4.2 VEHICLE DESCRIPTION SUMMARY

Detailed information on the Spacecraft 002 configuration is defined in drawings and process specifications of the "Spacecraft 002 Contractual End-Item Specification" (SID 63-699). Figure 4-1 is a diagram of Spacecraft 002 in the stack configuration.

The primary structures of Spacecraft 002 will be the launch escape subsystem, the spacecraft configuration command module, and the spacecraft configuration service module. Equipment for vehicle escape from the booster and for escape subsystem jettison from the command module will be contained in the launch escape subsystem. The command module will house the sequencer subsystem, retention and separation subsystem, earth landing subsystem, electrical power subsystem, and communication and instrumentation subsystem. The service module will be equipped with simulated reaction control subsystem quads, a blast shield, and an adapter mating ring facilitating service module installation with the Little Joe II booster.

The following are the model and general assembly drawings applicable to the Spacecraft 002 vehicle.

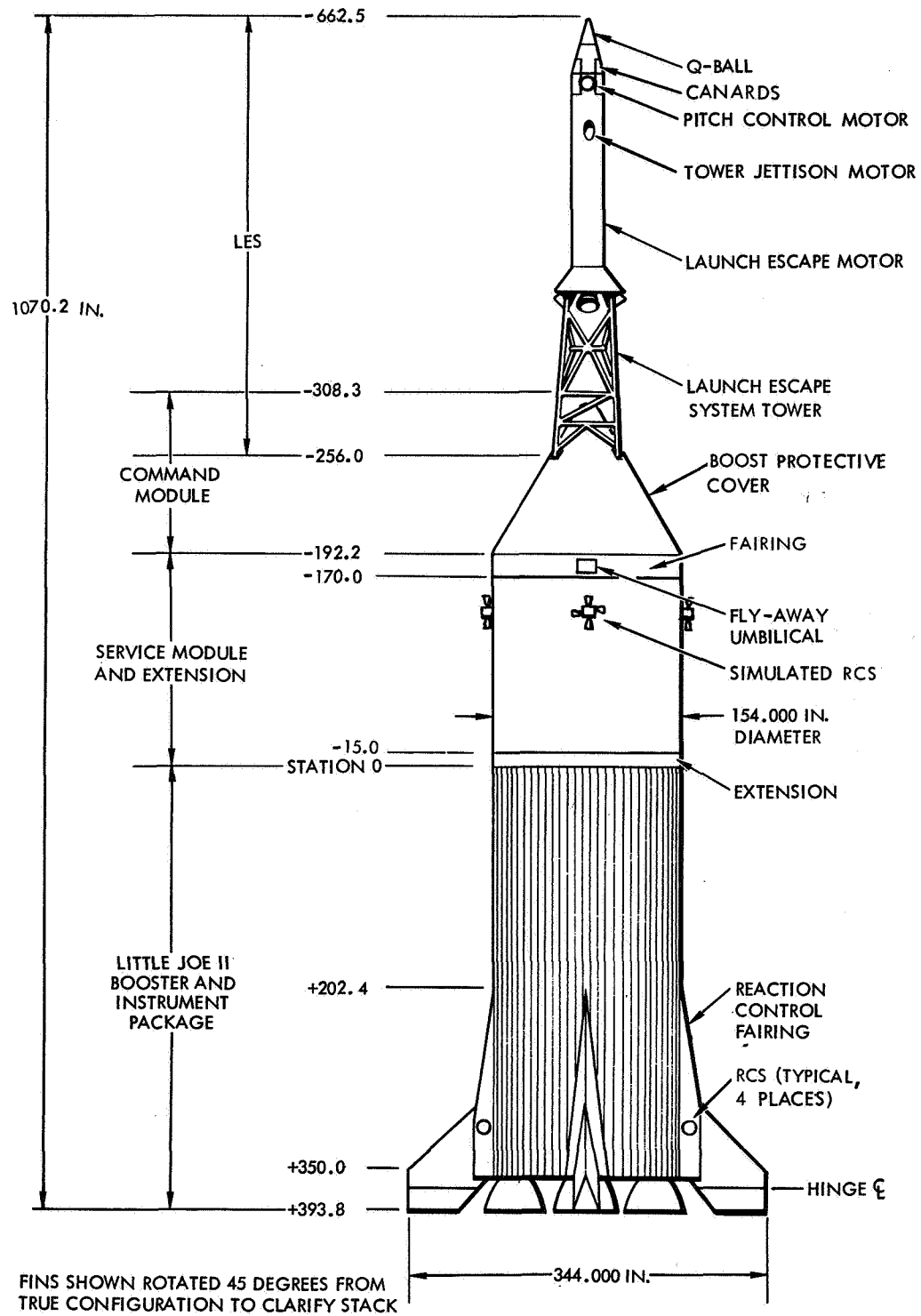


Figure 4-1. Spacecraft 002 Stack Configuration





End Item	Model Serial No.	General Assembly Drawing No.
Spacecraft stack	V14-3-002	V14-000002-121
Launch escape subsystem	V15-3-9	V15-000002-121
Command module	V16-3-2	V16-000002-121
Service module	V17-3-11	V17-000002-701
Earth landing subsystem	V16-3-2	V16-576003

#### 4.2.1 Launch Escape Subsystem

The launch escape subsystem for Spacecraft 002 will be a structural tower secured onto and over the apex of the command module and supporting a tubular casing enclosing the following components:

- Q-ball instrumentation package
- Pitch control motor
- Jettison motor
- Escape motor
- Canard airfoils
- Ballast

This configuration is shown on Figure 4-2. The boost protective cover for the command module is also considered a part of the launch escape subsystem. All components listed, including the tower, have been designated primary units of the subsystem. A diagram of LES components and subsystem orientation with the reference axis subsystem is presented on Figure 4-3.

##### 4.2.1.1 Q-ball Assembly

The Q-ball assembly constitutes the outermost tip of the launch escape subsystem which is most remote from the command module. It consists of transducers and associated attachments and wiring to sense air pressure and airflow direction through ports in the Q-ball surface. Angle of attack, angle of sideslip, and dynamic pressure will be derived from the measurements.

##### 4.2.1.2 Pitch Control Motor

Situated below the Q-ball is the solid-propellant pitch control motor to furnish a momentary thrust normal to the longitudinal axis of the launch escape subsystem to facilitate down-range escape during abort. However,

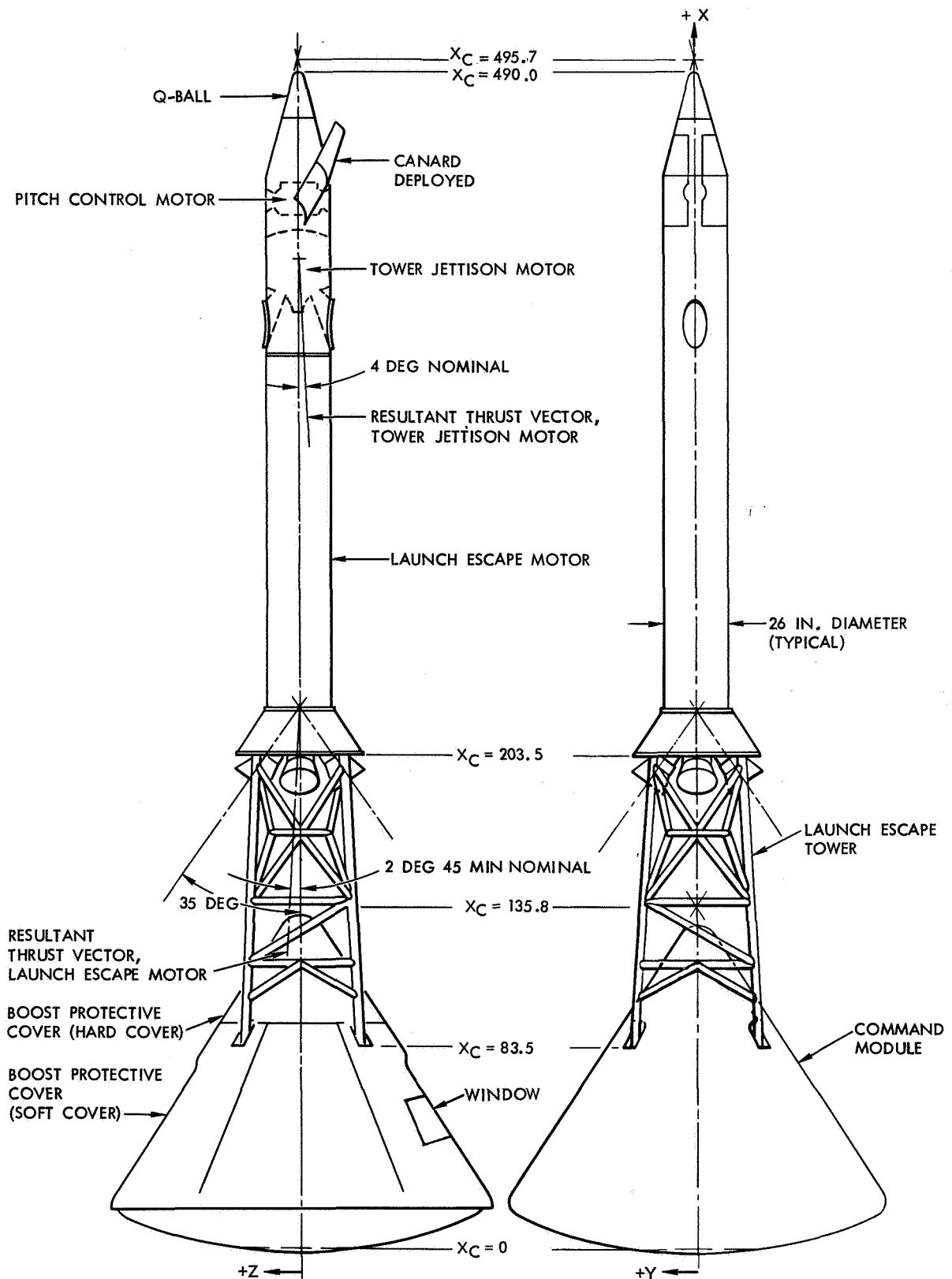


Figure 4-2. Launch Escape Vehicle Configuration

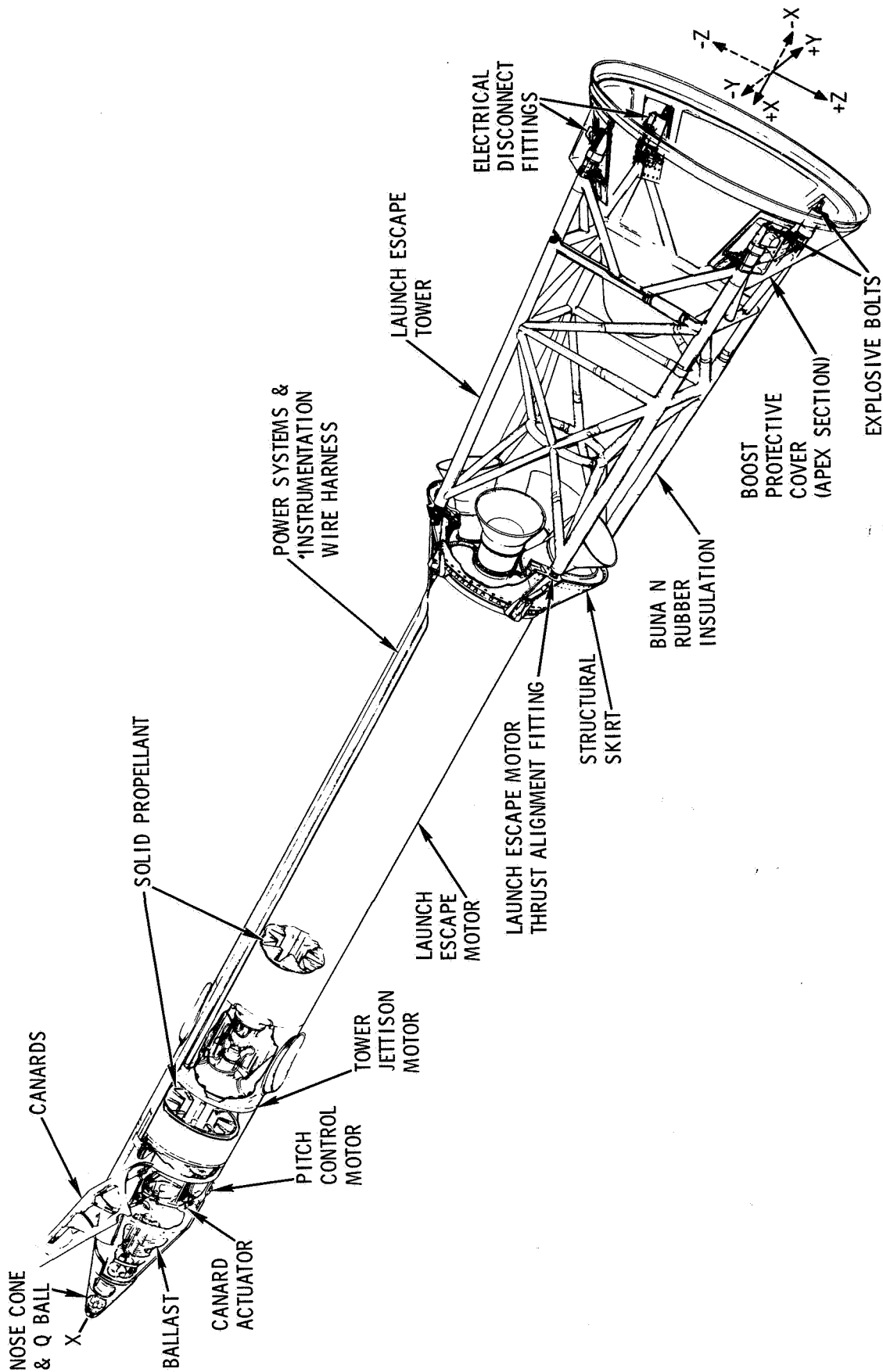
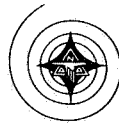


Figure 4-3. Launch Escape Subsystem Components

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its thrust has inconsequential effect in a power-on tumbling boundary abort. The pitch control motor is ignited with ignition of the launch escape motor at abort initiation. The pitch motor design thrust-time curve is contained on Figure 4-4.

#### 4.2.1.3 Jettison Motor

The tower jettison solid-propellant motor propels the entire launch escape subsystem plus the boost protective cover away from the command module preparatory to parachute deployment for recovery. Thrust through two canted exhaust nozzles powers the launch escape subsystem on a curving trajectory. The motor design thrust-time curve is contained on Figure 4-5.

#### 4.2.1.4 Escape Motor

The launch escape solid-propellant motor generates thrust through four canted exhaust nozzles to propel the launch escape vehicle on an escape course away from the booster. The motor design thrust-time curve is contained on Figure 4-6.

#### 4.2.1.5 Canards

The canards constitute a part of the launch escape subsystem and consist of two deployable surfaces and an actuating and locking mechanism located in the LES just below the Q-ball. The canard surfaces in the stowed position form a portion of the LES upper cylindrical casing. They act as airfoils upon deployment at 11 seconds after abort initiation to subsequently orient and stabilize the launch escape vehicle aft heat shield toward earth for recovery.

#### 4.2.1.6 Escape Tower

The launch escape tower is an open frame structure of welded titanium tubing in the form of a truncated rectangular pyramid. It is the intermediate structure between the command module and the escape subsystem rocket motors. An ablative Thermolag material is applied onto the tower to minimize aerodynamic heating during boost and to minimize rocket exhaust heating during abort. A structural skirt facilitates launch escape motor attachment to the upper base of the tower. The attachments are adjustable for thrust alignment. The tower base is secured to the command module at four points by quick release dual-mode exploding bolt mechanisms.

#### 4.2.1.7 Boost Protective Cover

The boost protective cover encloses the entire conical surface of the command module to protect it from aerodynamic heating during high/Mach

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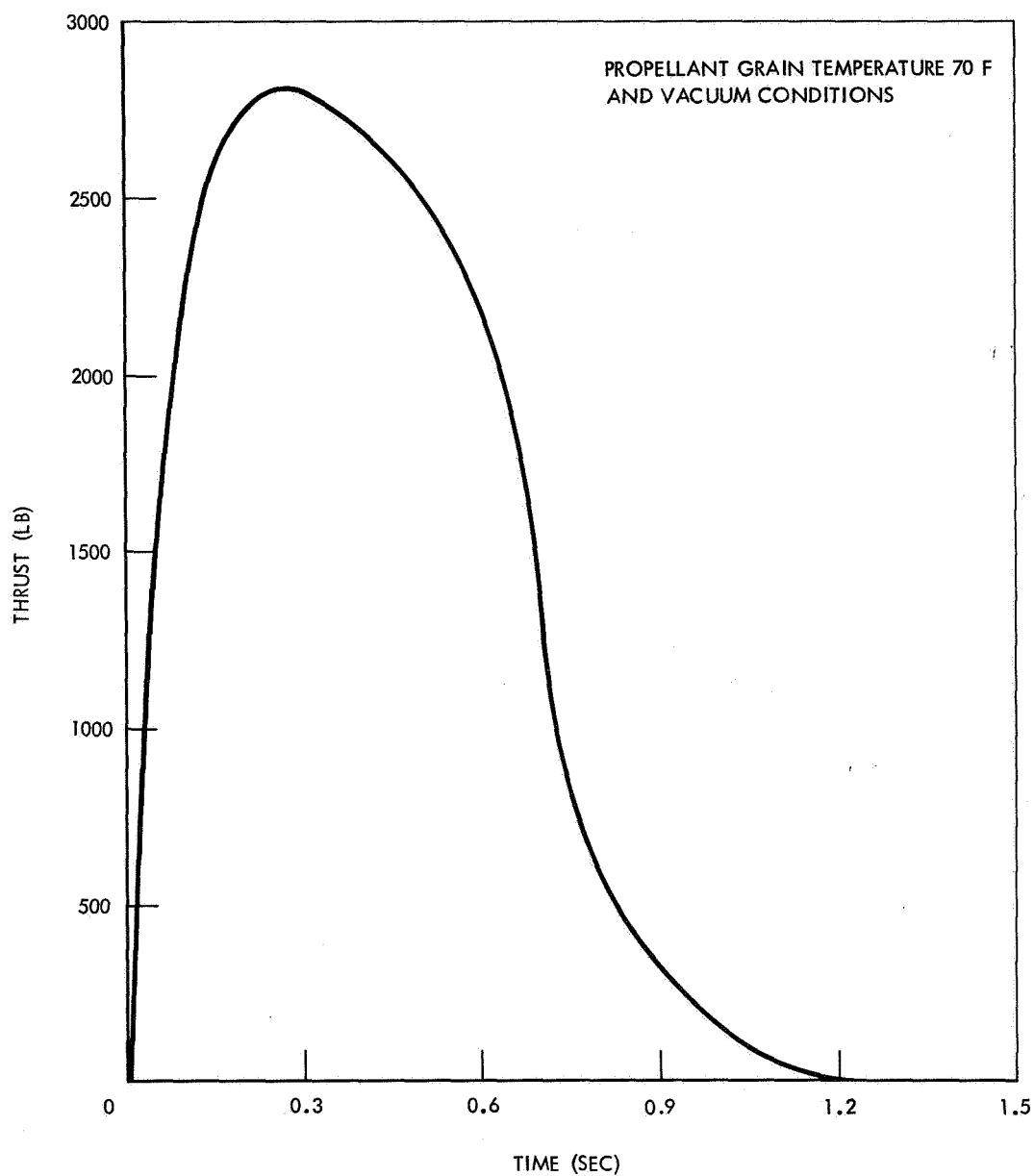
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Figure 4-4. Typical Thrust-Time Curve of Pitch Control Motor

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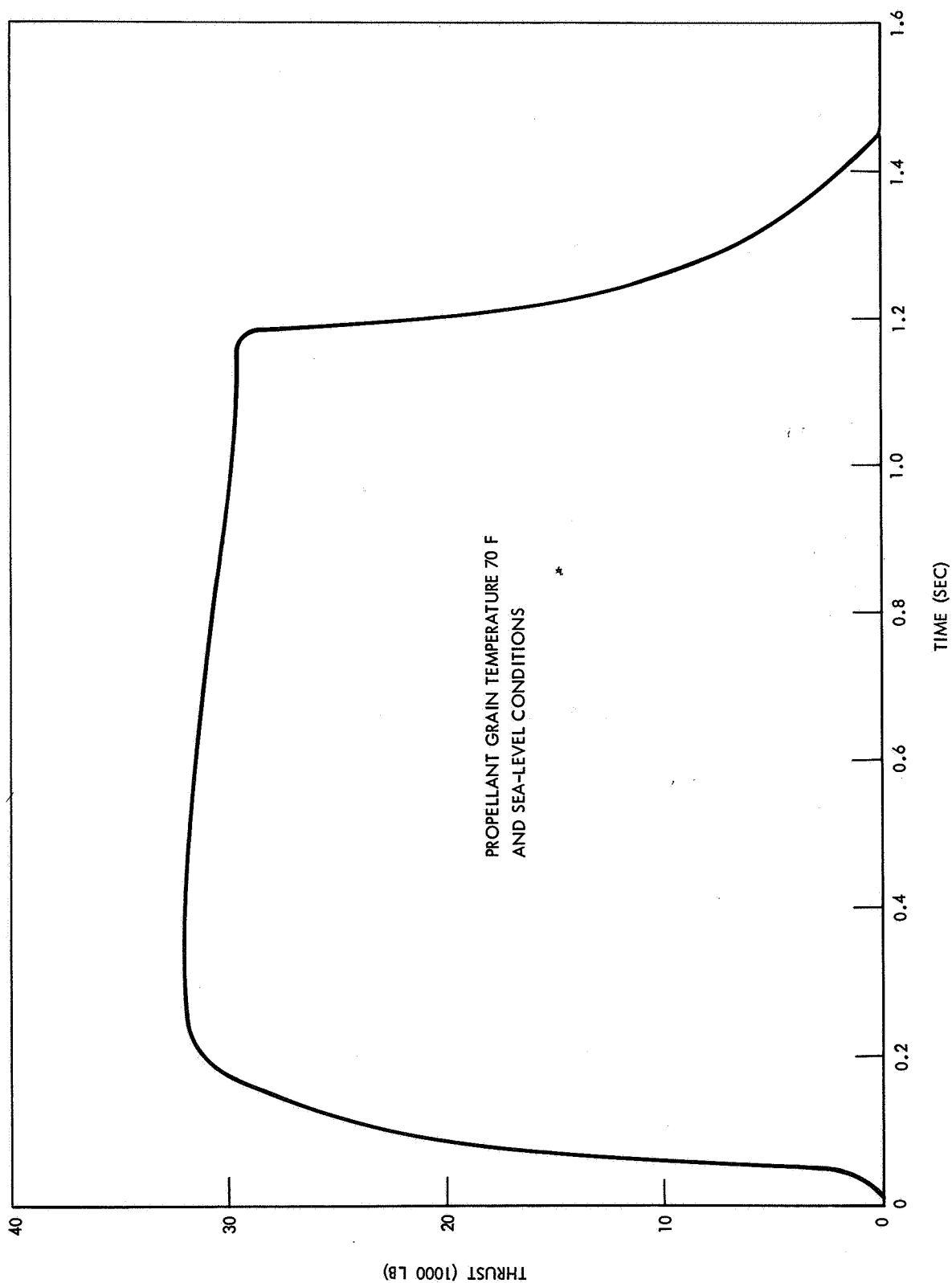
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Figure 4-5. Typical Thrust-Time Curve of Tower Jettison Motor

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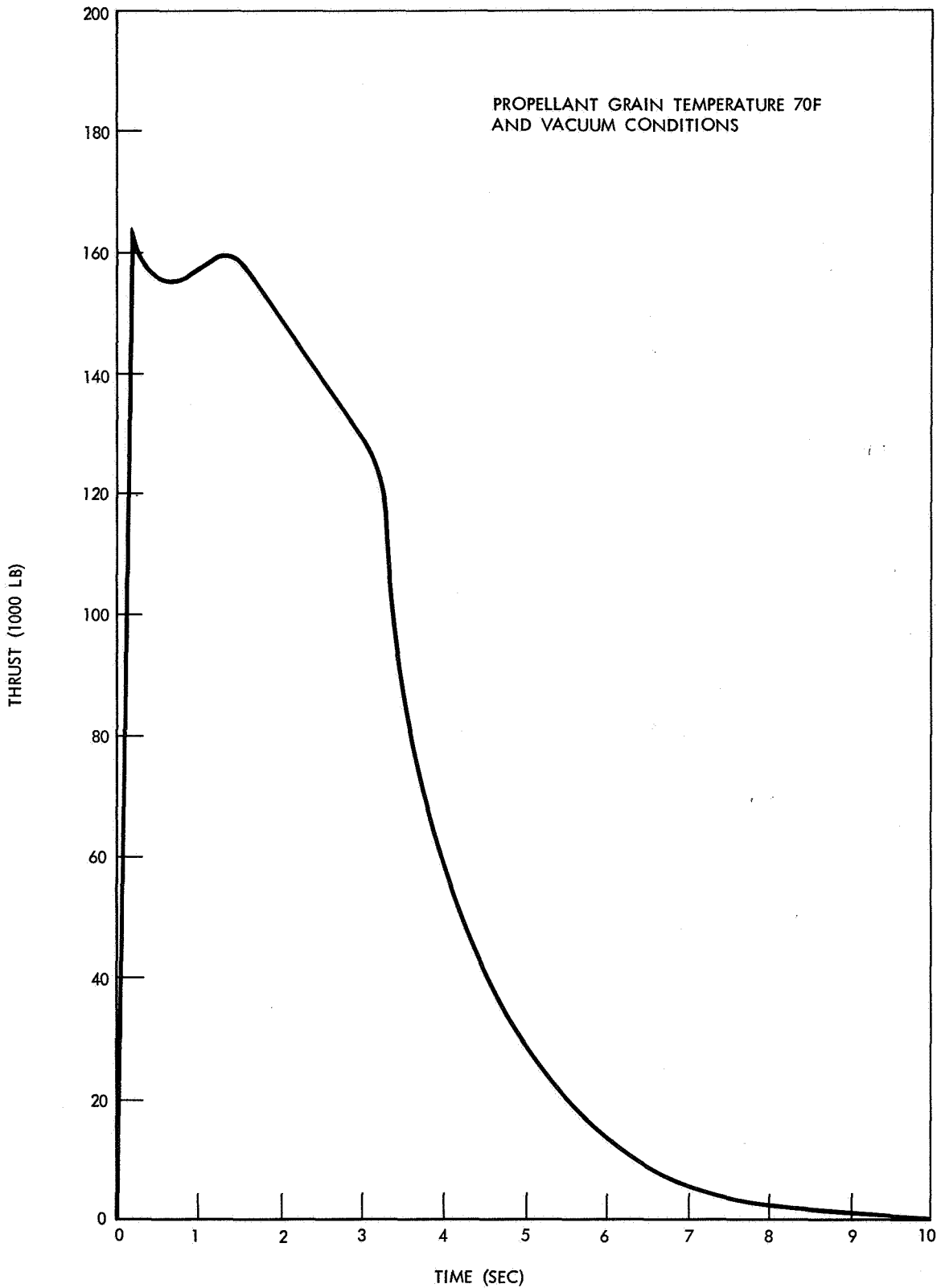
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Figure 4-6. Typical Thrust-Time Curve of Launch Escape Motor

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number-high dynamic pressure launch. The soft portion of the cover is fabricated in seven sections. The forward part of the cover is of hard honeycomb fiberglass with an outer ablator while the aft portion is of Teflon-impregnated soft fiberglass cloth with an ablative surface of cork. The cover will be pulled off the command module by the tower during launch escape subsystem jettison.

#### 4.2.2 Earth Landing Subsystem

The earth landing subsystem comprises the drogue parachute subsystem, the pilot parachute subsystem, and the main parachute subsystem.

##### 4.2.2.1 Drogue Parachute Subsystem

Two 13.7-foot diameter fist ribbon nylon parachutes and two deployment mortars constitute major parts of this subsystem. Mortar firing deployment of both drogue chutes occurs two seconds after LES jettison. Both chutes will be in the 40-percent reefed configuration to minimize opening shock until disreefing occurs eight seconds after line stretch. Both drogues will be released from the command module as the vehicle descends through 11,000 feet attitude.

##### 4.2.2.2 Pilot Parachute Subsystem

Three 7.2-foot diameter ring slot parachutes and three deployment mortars constitute the major parts of this subsystem. Mortar firing deployment of all three pilot chutes occurs with the 11,000-foot altitude parallel signal that releases the two drogue chutes. Each pilot chute pulls one main parachute bag overboard to full extension of the main chute risers before main chute deployment. The pilot chutes remain attached to the main chutes.

##### 4.2.2.3 Main Parachute Subsystem

Three 83.5-foot diameter open-ring ringsail parachutes constitute this subsystem. Each main chute is deployed in the nine percent reefed configuration to minimize opening shock. Disreefing occurs eight seconds after line stretch.

#### 4.2.3 Primary Structures

The command module, the service module, and the service module Little Joe II adapter for interface with the Little Joe II booster comprise the primary structure of Spacecraft 002.





#### 4.2.3.1 Command Module Structure

The command module is constructed of a constant-depth, brazed-honeycomb, steel outer shell and a bonded-aluminum honeycomb inner structure. The outer shell contains a layer of cork coating for crew compartment thermal protection. The cork simulates the spacecraft ablative material. The outer shell is divided into three major sections: the forward compartment heat shield, the crew compartment heat shield, and the aft compartment heat shield. The forward heat shield is a complete conical unit secured to the inner structure at four points. The crew compartment heat shield contains numerous cutouts for access panels, windows, and the crew hatch. The aft heat shield is a continuous unit secured to the inner structure by tension bolts. The booster thrust is transmitted directly through this heat shield to the inner structure.

The inner structure is the load-bearing article of the command module; it contains a forward bulkhead, forward and aft sidewalls, and aft bulkhead. Four continuous longerons support the launch escape subsystem at the forward end, and these plus additional longerons in the aft side wall distribute the launch booster thrust load. Four structural beams attached to the forward bulkhead distribute the internal pressure, forward heat shield ejection load, drogue and pilot chute mortar loads, and main parachute attachment loads. The numerous cutouts correspond to those in the outer shell.

Both the outer shell and the inner structure are symmetrical about a common centerline.

#### 4.2.3.2 Service Module Structure

The Spacecraft 002 service module will be a spacecraft configuration service module structural shell only. Command and service module separation apparatus, signal distribution units, tape modulation packages, transducers, and amplifiers will be the only equipment contained within the service module. The shell is constructed of one-inch aluminum honeycomb in six segments. Its diameter is 154 inches, and it is 155 inches long. Nonfiring simulated reaction control subsystem (RCS) engine quads will be installed on the exterior surface of the service module.

#### 4.2.3.3 Service Module - Little Joe II Adapter

A 15-inch length extension adapter shell interfaces with the service module at one end and with the Little Joe II booster at the other in order to achieve the necessary dimensional transition. A pressure barrier mounted



across the adapter inside diameter shields the command module from Little Joe II destruct blast pressure, should destruction become necessary.

#### 4.2.4 Retention and Separation Subsystem

Spacecraft 002 will contain the capability for retention and separation of (1) the command module and the service module, (2) the launch escape subsystem and the command module, and (3) the forward heat shield and the command module.

##### 4.2.4.1 CM-SM Retention and Separation

Three tension tie rods, each connected to a mounting plate and situated approximately 120 degrees apart around the periphery of the command module base, secure the command module to the service module structure. Each tie rod contains two shaped explosive charges that detonate together to fracture the rod (detonation of a single charge is sufficient to fracture the rod) and together with severance of the electrical umbilical and plumbing hard lines between the command module and service module by cutters, the command module is completely disjoined from the service module and free to be separated by launch escape motor thrust.

##### 4.2.4.2 LES-CM Retention and Separation

The launch escape subsystem is secured to the command module through the escape tower at its four extremities by dual-mode bolts. Bolt fracture by self-contained redundant explosive charges is accomplished to free the entire launch escape subsystem and the boost protective cover for separation by the jettison motor.

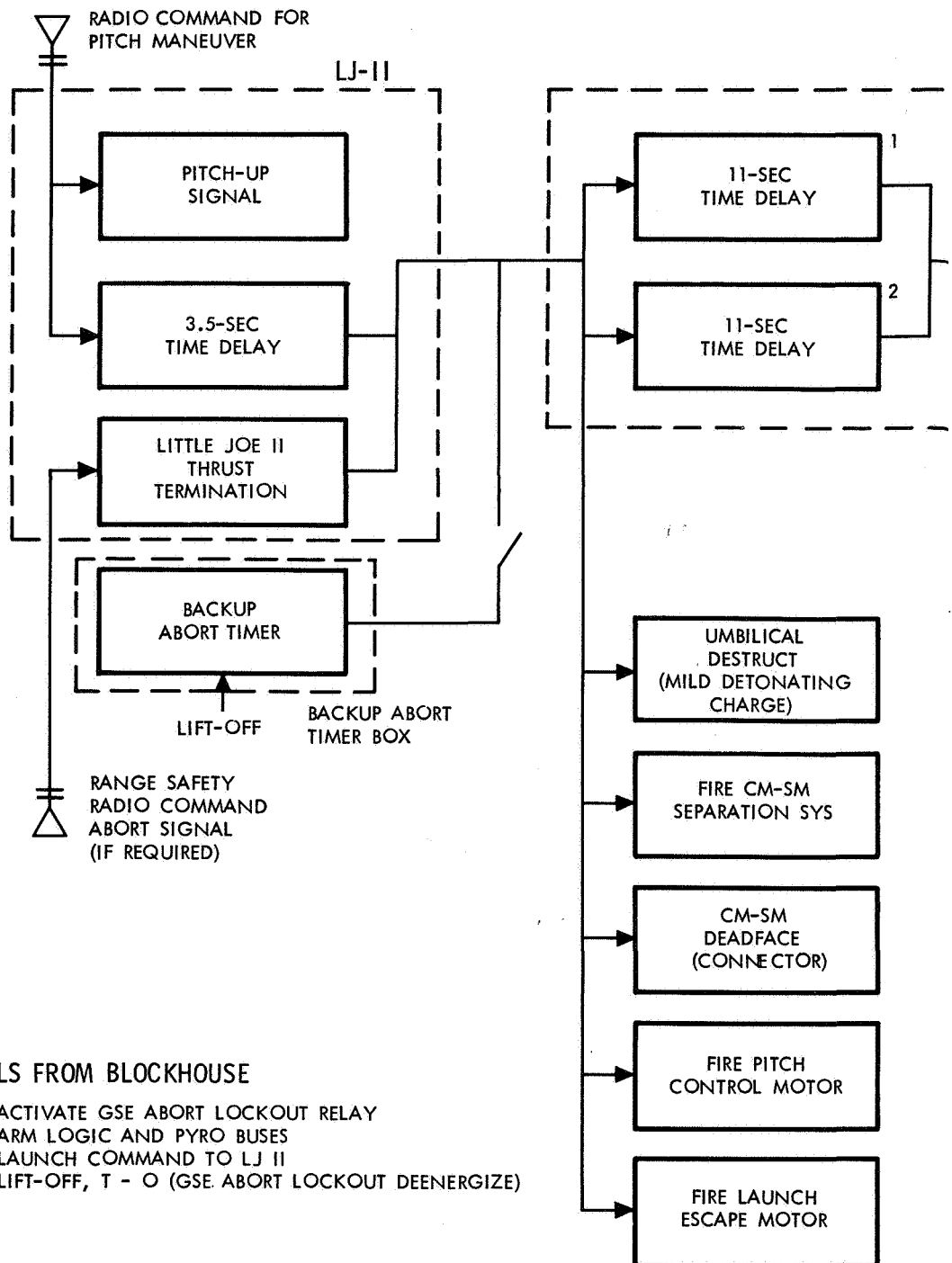
##### 4.2.4.3 Forward Heat Shield—Command Module Retention and Separation

Four tension tie rods, each connected to a thruster-ejector assembly, secure the forward heat shield to the command module. The four thruster-ejectors are situated around the exterior of the forward egress tunnel at approximately 90 degrees apart. Two gas generators operate the thrusters, each generator operating two thrusters 180 degrees apart. Thruster firing occurs 0.4 seconds after jettison motor ignition, fracturing the tension tie rods and ejecting the heat shield from the command module.

#### 4.2.5 Sequencer Subsystem

Events of power-on tumbling boundary abort mission A-004 will be scheduled by the mission sequencer, the ELS sequencer, and the tower sequencer. A block diagram of the sequencer subsystem is presented on Figure 4-7.

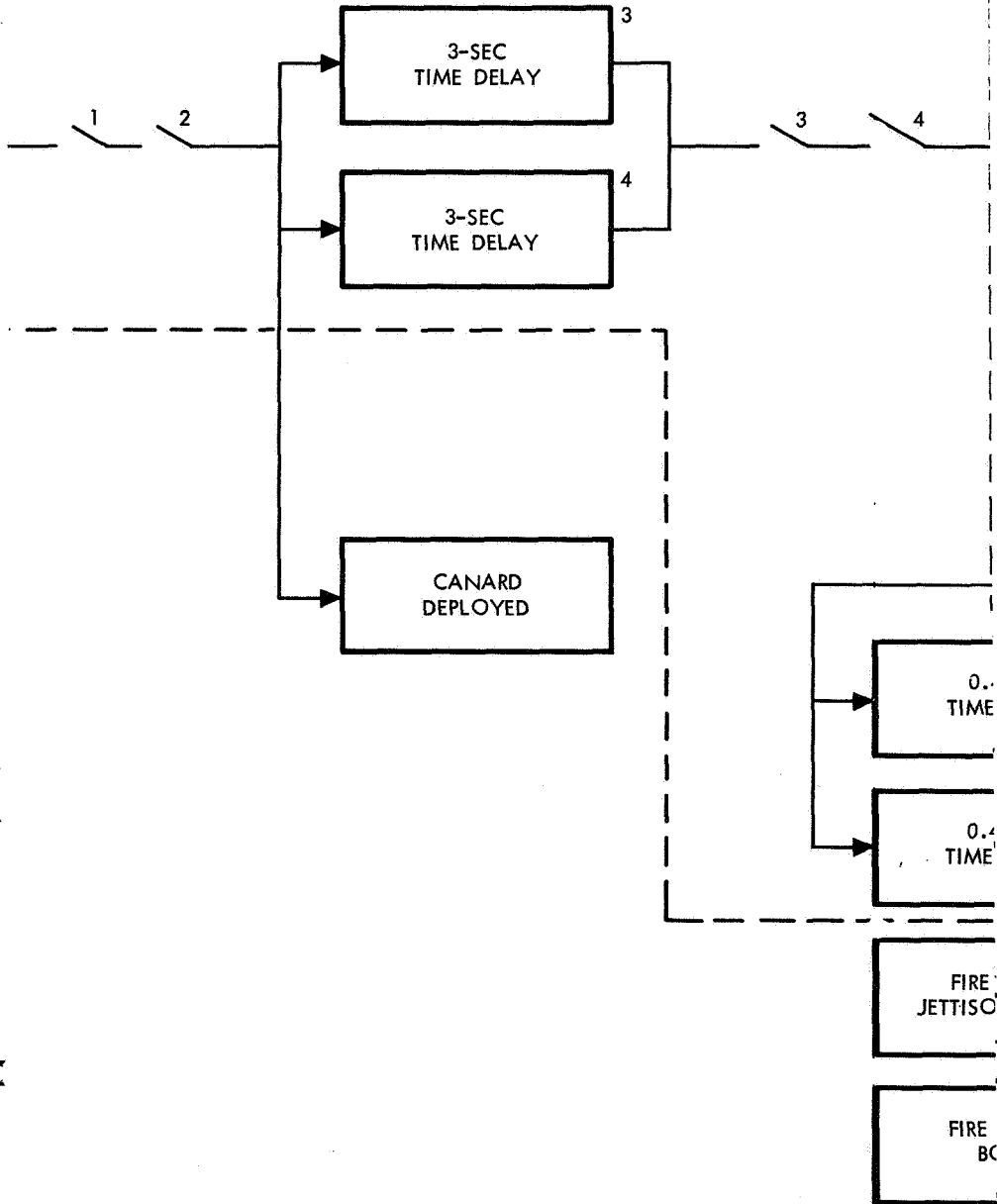


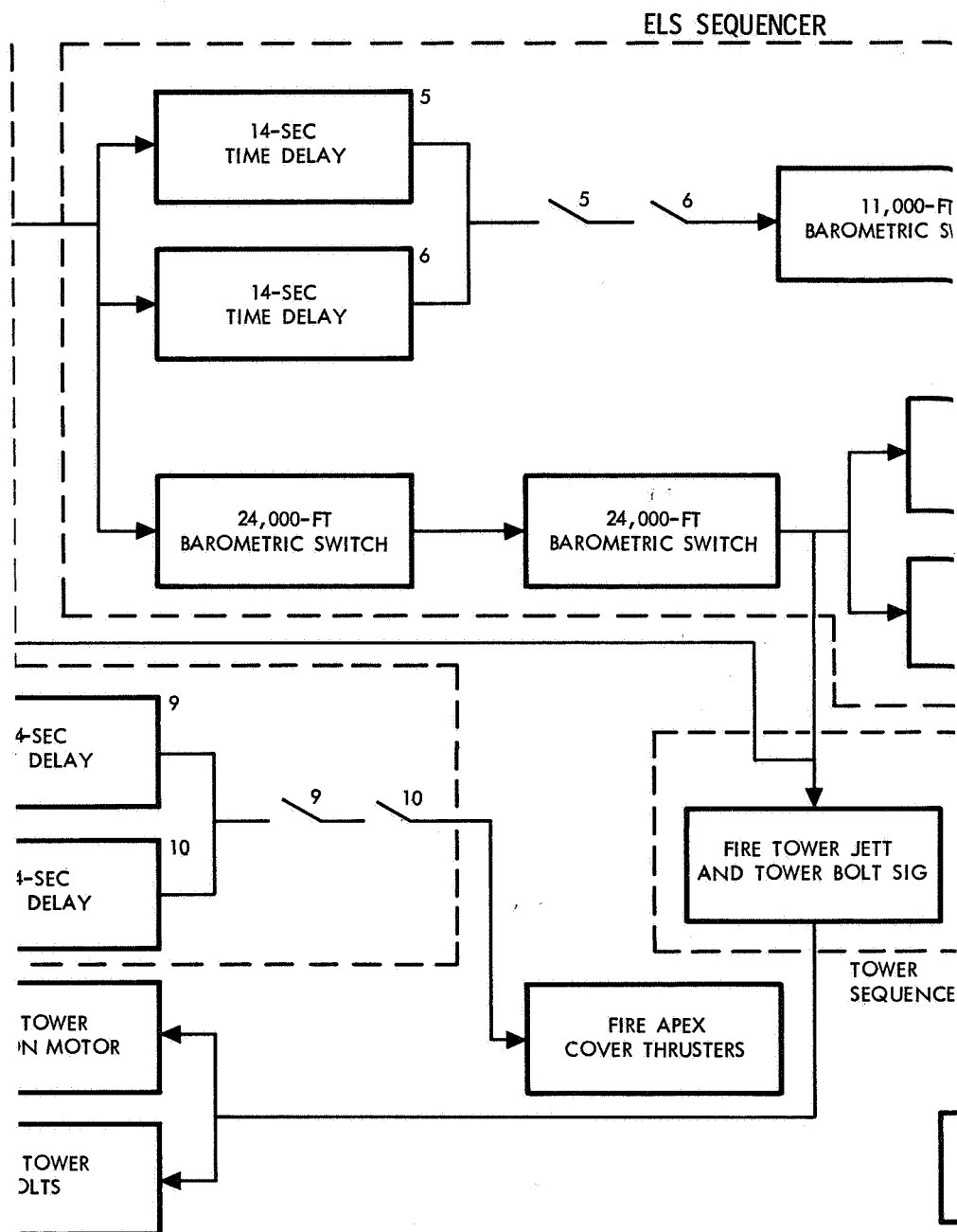


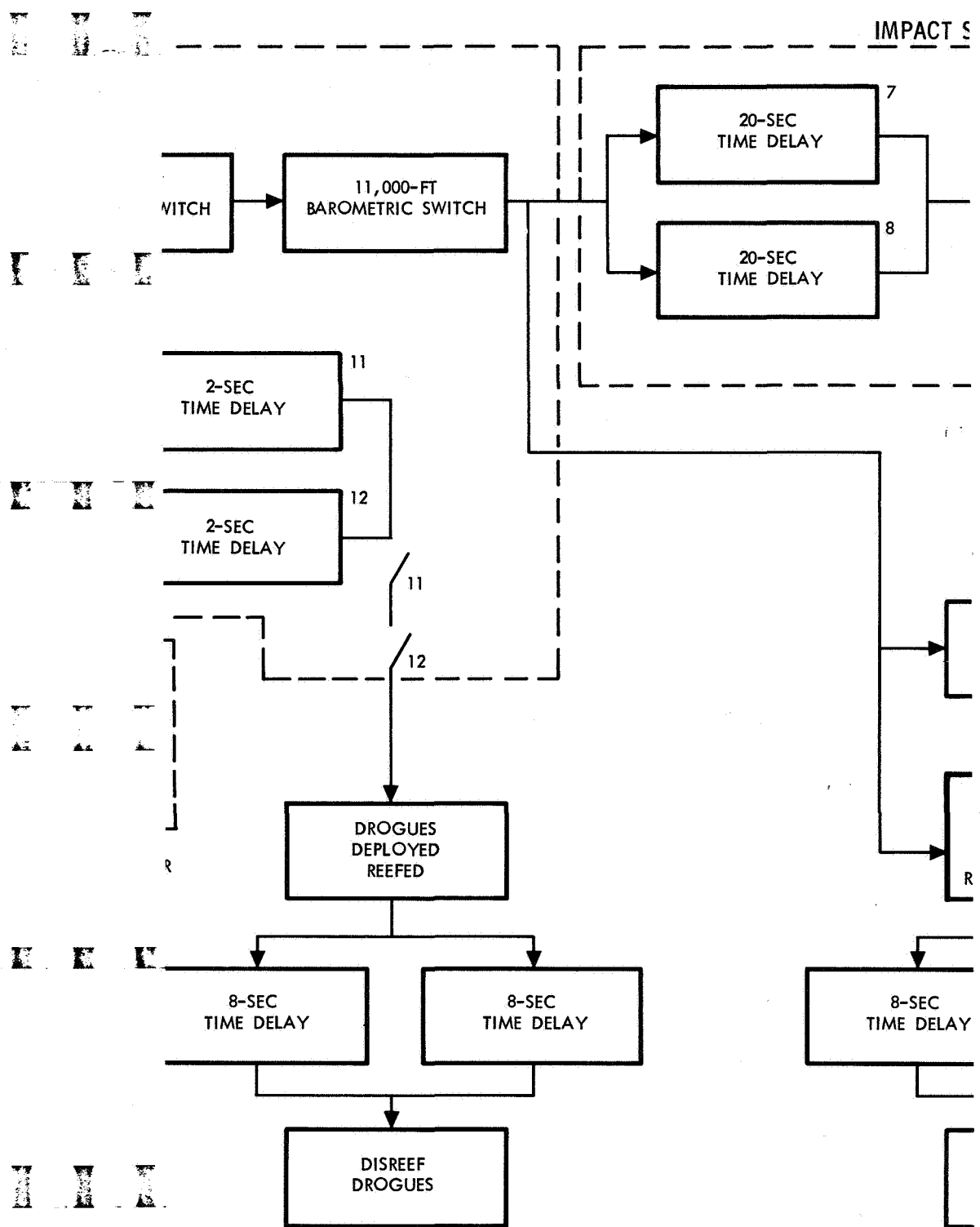
#### SIGNALS FROM BLOCKHOUSE

1. ACTIVATE GSE ABORT LOCKOUT RELAY  
ARM LOGIC AND PYRO BUSES
2. LAUNCH COMMAND TO LJ II
3. LIFT-OFF, T - O (GSE ABORT LOCKOUT DEENERGIZE)

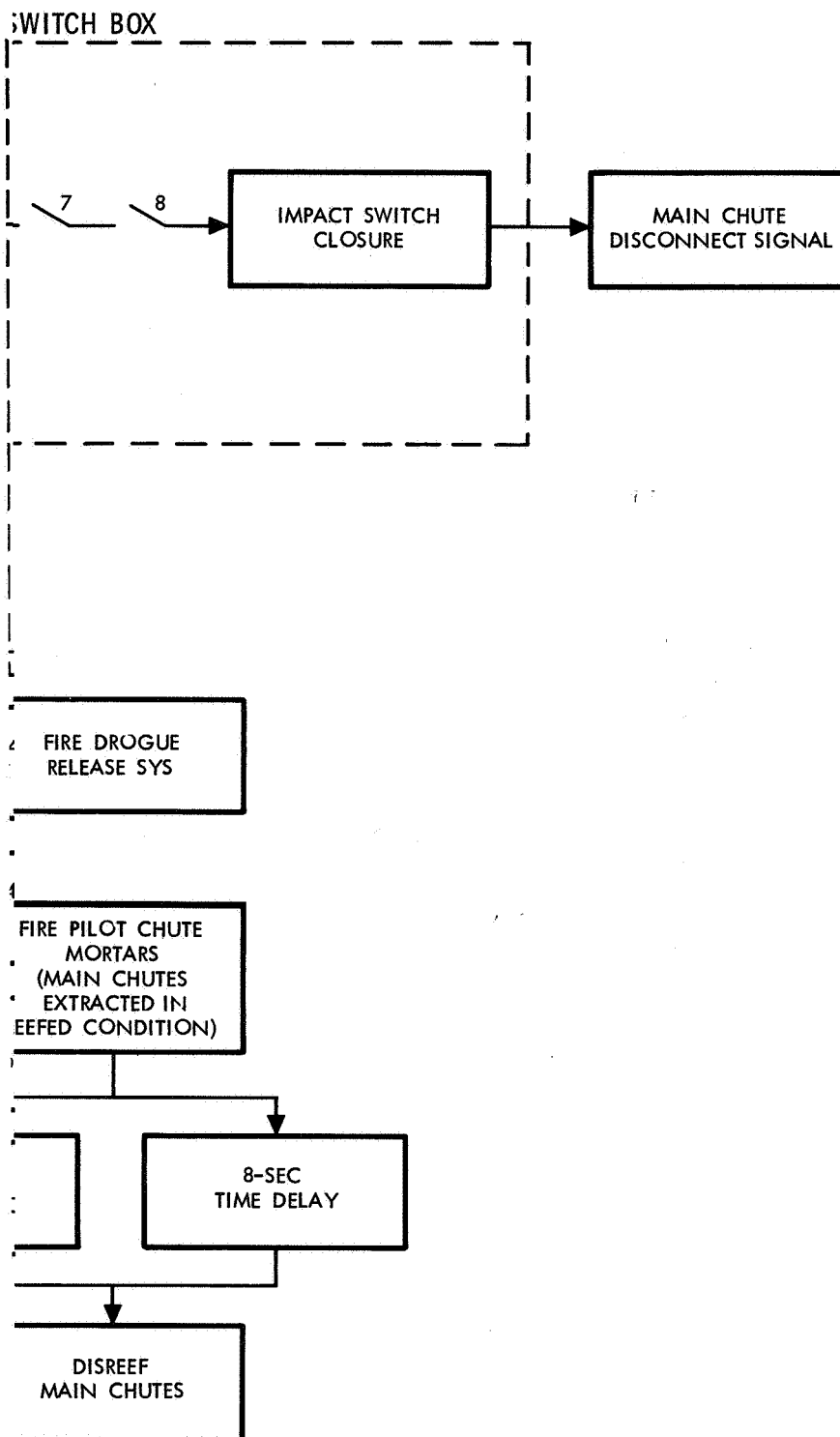
MISSION SEQUENCER







Figure



#### 4-7. Mission Sequencer Subsystem





#### 4.2.5.1 Mission Sequencer

The mission sequencer furnishes the signals for command module separation from the service module, canard deployment, forward heat shield thruster firing, and safing and arming of the tower and ELS sequencers. An abort backup timer constitutes a part of the mission sequencer, but it will be shunted nonfunctional for Mission A-004. The timer will be of the same configuration as that on Boilerplate 22.

#### 4.2.5.2 Tower Sequencer

The tower sequencer is situated in the command module on Spacecraft 002. It consists of relays which receive signals from the mission sequencer and, upon closure, allows electrical power from pyrotechnic batteries to energize squibs and igniters for firing the launch escape, pitch control, and jettison motors, and for detonating the four dual-mode explosive bolts that secure the LES structure to the command module.

#### 4.2.5.3 ELS Sequencer

This component consists of two redundant packages located in the command module right-hand equipment compartment. Each package contains two baroswitches, one for circuit completion to the drogue chute mortars and the other for circuit completion to the pilot chute mortars plus drogue chute disconnect mechanism.

#### 4.2.6 Electrical Power Subsystem

Six Eagle-Picher silver oxide-zinc storage batteries comprise the onboard electrical power source for Spacecraft 002. Two batteries (type MAP 4095) will power the instrumentation through the NASA-supplied power control box as shown on Figure 4-8. As indicated on this Figure, another two batteries (type MAR 4090) will furnish power to the sequencer subsystem logic circuit, and the third pair of batteries (type ME 461-0007) will serve as the pyrotechnic power source.

#### 4.2.7 Communication and Instrumentation Subsystem

##### 4.2.7.1 Telemeter and Antenna

Two telemeter transmitters will transmit flight data through a multiplexer and one VHF antenna. This antenna is one of two scimitar antennas. The other will be nonfunctional on this vehicle.

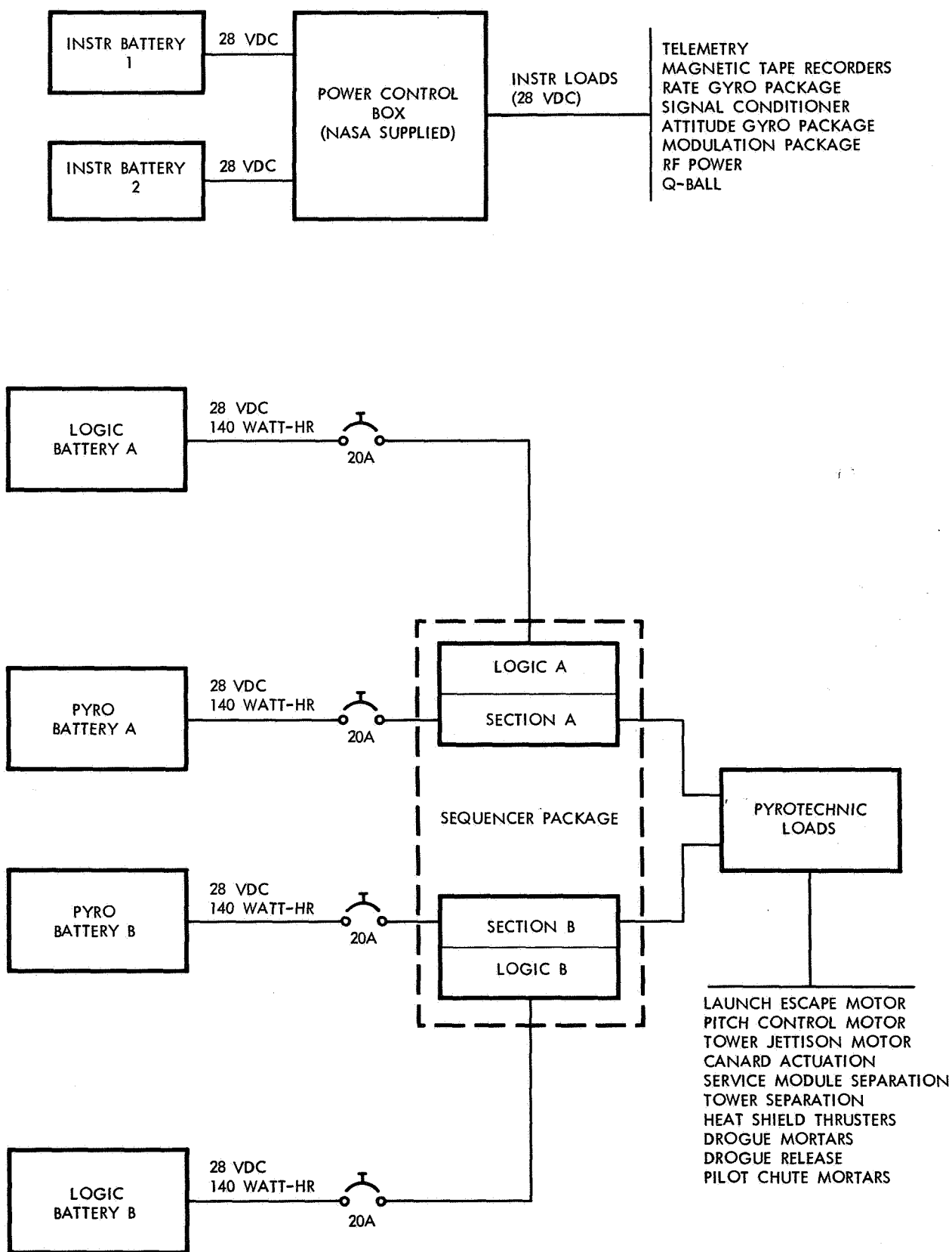


Figure 4-8. Electrical Power Distribution



#### 4.2.7.2 Onboard Recorder

Two 14-channel tape recorders, including recorder electronics and tape transport, will be onboard to record parameters requiring high-frequency response. One of the two recorders will be operating in conjunction with ten tape modulation packages.

#### 4.2.7.3 End Instrumentation and Signal Conditioners

Accelerometers, current monitors, position transducers, rate transducers, strain gauges, temperature transducers, and acoustic transducers make up the end instrumentation. Bridge adjust units, thermocouple compensation, and phase-sensitive demodulators will constitute the signal conditioning equipment on this vehicle.

#### 4.2.7.4 C-Band Transponder and Antenna

Two C-band transponders will be onboard to permit accurate radar tracking. Each transponder will respond independently to incoming pulse code signals and reply independently to tracking stations through two helix antennas.

#### 4.2.8 Launch Vehicle

The Little Joe II launch vehicle will be an airframe comprising a two-section cylindrical body containing four Algol solid-propellant rocket motors, a set of airfoil fin-elevons, a reaction control subsystem, and timer units (command control logic, etc.). Two of the Algol rockets will be fired for liftoff; the remaining will be ignited just prior to burnout of the first pair. Four fins and their associated hydraulically actuated elevon control surfaces will be spaced at equal distances around the afterbody to provide the primary mode of vehicle stability and attitude control. The hydrogen peroxide monopropellant reaction control subsystem will provide stability and attitude control assistance. Two timer units will be onboard the Little Joe II: one unit for providing the 3.5-second delay from pitch-up initiation to LEV abort initiation (staging timer) and the other unit for backup initiation of pitch-up based on elapsed time from lift-off in the event the booster fails to respond to the radio command for pitch-up (RCS timer).

Detailed information of the launch vehicle and its performance will be contained in "Performance and Interface Specification, Apollo Spacecraft 002 and Little Joe II Launch Vehicle" (SID 63-949).



#### 4.2.9 Configuration Differences

Configuration differences of Spacecraft 002 in relation to the abort Mission A-003 vehicle (Boilerplate 22) are as follows:

Article	Boilerplate 22	Spacecraft 002
Command module	Boilerplate	Spacecraft
Service module	Boilerplate	Spacecraft
Boost protective cover	Eight-section (boilerplate)	Seven-section (spacecraft)
CM-SM separation	Compression pads and radial beams	Compression pads and radial beams plus electrical umbilical and plumb hard lines

Configuration differences between Apollo vehicles of successive launch escape subsystem abort missions in the WSMR Apollo flight program are summarized on Figure C-2 of Appendix C.

#### 4.3 MASS PROPERTIES DATA

The preliminary mass properties data currently available on Spacecraft 002 are presented on Table 4-1. The values are those currently proposed for the vehicle in the flight readiness state on the launch pad. More complete mass properties data will be available at a later date.

Table 4-1. Weight, Gravity, and Inertia Summary

Article	Weight (lb)	Center of Gravity			Moment of Inertia (Slug Ft <sup>2</sup> )		
		X	Y	Z	Roll	Pitch	Yaw
Launch escape system	8,200	1301.5	0.0	-0.1	528.0	20,446.2	20,445.8
Command module	11,000	1037.2	0.6	5.4	5284.9	4740.2	4488.6
Service module	5,000	905.2	-1.2	-11.8	5521.7	7185.7	6600.0
Service module adapter (extension)	1,000	828.5	-0.8	0.1	1245.9	626.0	626.0



## 5.0 FLIGHT CONSTRAINTS

Flight constraints are defined as the minimum mandatory testing which must be satisfactorily accomplished prior to execution of the mission under consideration. Constraints are "hard" requirements, not arbitrarily removed, and, if not accomplished, will hold the mission under consideration until accomplishments has been achieved. The constraints on Spacecraft 002 are described in Table 5-1.



Table 5-1. Flight Constraints

Item	Constraint	Procedure	Reason
Structural system CM and SM	Satisfactory completion of combined command and service module limit load static tests at selected test conditions	ATR 600 (Airframe 004)	Spacecraft 002 is the first abort vehicle with a spacecraft command and service module structure.
Ordnance system, CM-SM umbilical separation system	Satisfactory completion of CM-SM umbilical pyrotechnic disconnect tests	ATR 100A	Spacecraft 002 is the first abort vehicle equipped with a spacecraft CM-SM umbilical and separation system.
Mission A-003 (Boilerplate 22)	Satisfactory completion of the following second-order test objectives:  1. Demonstrate orientation of the LEV to a main heat shield forward attitude  2. Determine the damping of the LEV oscillations with the canard subsystem deployed	NASA mission directive for mission A-003	Mission A-003 is the first mission during which the canards will orient and stabilize the CM from high-altitude tumbling.
Mission A-003 (Boilerplate 22)	Satisfactory performance of dual-mode bolts that secure the launch escape tower to the CM	NASA mission directive for Mission A-003	Mission A-003 is the first mission during which the dual-mode bolts free the LES from the CM.



## 6.0 MISSION RULES SUMMARY

The mission rules contain information both on the launch limitations that govern continuation or suspension of launch countdown and on the design test point conditions of the launch escape vehicle. Launch site wind velocities and directions, visibility, and launch angle are the elements included under launch limitations. Mach number, dynamic pressure, flight path angle, angle of attack, and pitch rates constitute the escape vehicle test point conditions at start of abort. Launch limitations must not be exceeded to allow the launch vehicle the greatest probability of attaining the design point conditions within the allowable dispersions.

The limitations and test point conditions are defined in Appendix B of this document.

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## 7.0 SCHEDULE

The various phases of Spacecraft 002 activity will be accomplished in accordance with the schedule presented on Figure 7-1. This information is based on Master Development Schedule (MDS) 8, Revision 2, dated 22 January 1965.

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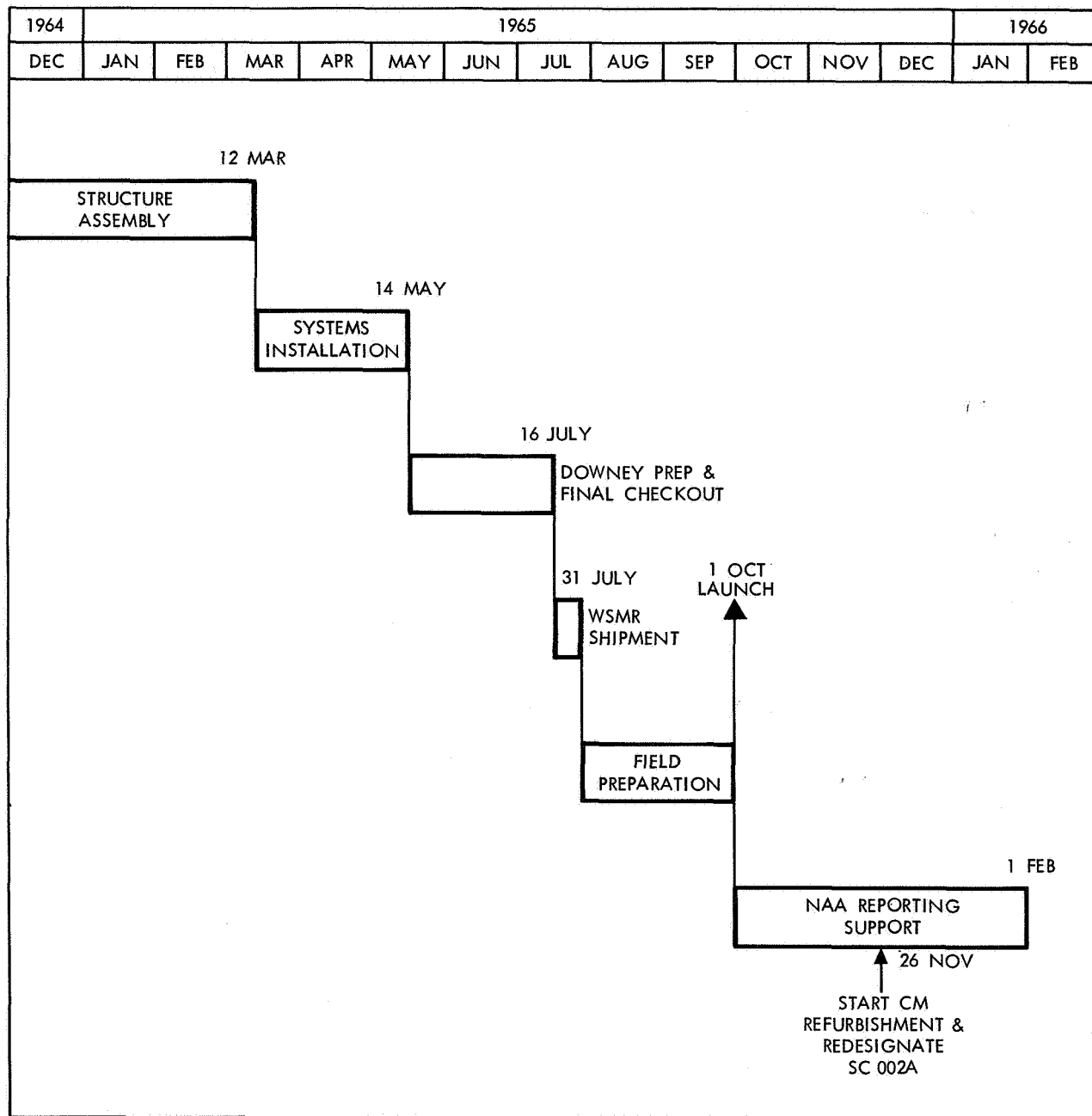


Figure 7-1. Activity Schedule

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## 8.0 VEHICLE CHECKOUT PLAN

### 8.1 VEHICLE PREPARATION (S&ID DOWNEY)

Upon completion of systems installation by Manufacturing, the Spacecraft 002 vehicle will be delivered to NAA Apollo Test and Operations for test preparation at Downey prior to shipment to WSMR. Preparation will consist of five phases. Figure 8-1 defines each phase and the sequence of tests to be performed on Spacecraft 002 from the completion of systems installation until shipment to WSMR. The block diagram in Figure 8-2 traces the proposed flow movement of the major components of the vehicle through the Downey complex. Upon completion of system installation, the following will be accomplished in Building 290: mechanical fit checks, weight and balance, configuration verification, and open work items. The power-on testing sequence will be conducted in Building 001 and in the high-bay assembly tower.

#### 8.1.1 First Phase

In Building 290, a complete examination of the vehicle systems and subsystems will be performed to verify conformance with the latest configuration, specifications, and engineering orders to ensure the readiness of the hardware for checkout operations. Parts shortages will be remedied, and R & D equipment that was not available in the systems installation area will be installed. During this period, the essential flight hardware items will be installed in the command module and service module, and the preliminary weight and balance will be accomplished. These units will then be mated on the stacking fixture, H14-9030, and alignments will be completed. The boost protective cover will be fit checked on the command module with the launch escape tower installed. While these units are stacked, a configuration verification and an inspection shakedown will be performed. The vehicle will be demated, and open work items will be completed. The vehicle will be prepared for movement to Building 001.

#### 8.1.2 Second Phase

The command module, with the apex cover removed, will be mated to the service module H14-9030 combination in Building 450 (Navajo tower). The launch escape tower will be mated to the command module, and the canard and the Q-ball will be electrically connected to the launch escape tower skirt. During this period, the GSE will be assembled and mated with spacecraft special test equipment (STE) principally meter room equipment, and an integrated GSE systems test will be performed. The STE will present



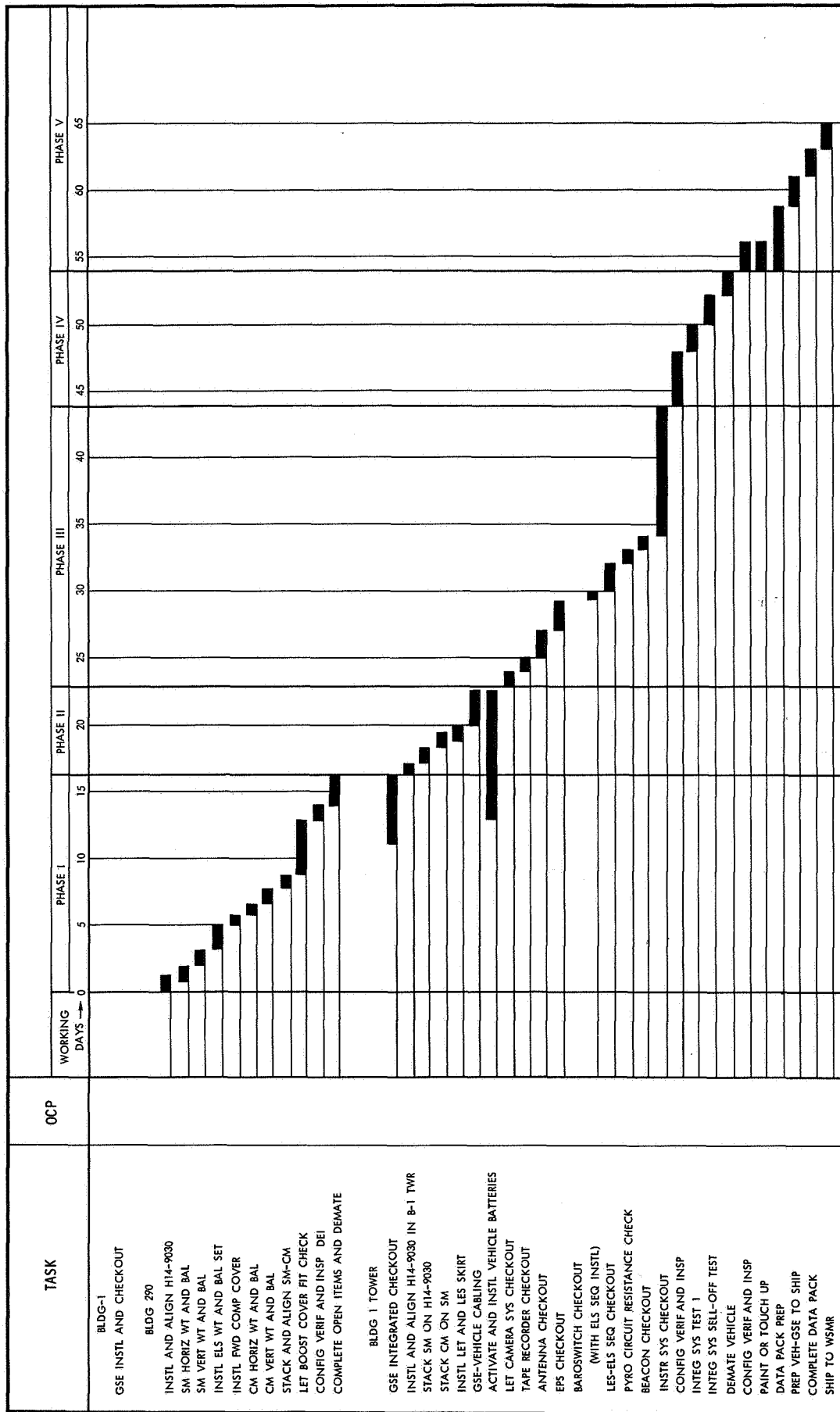
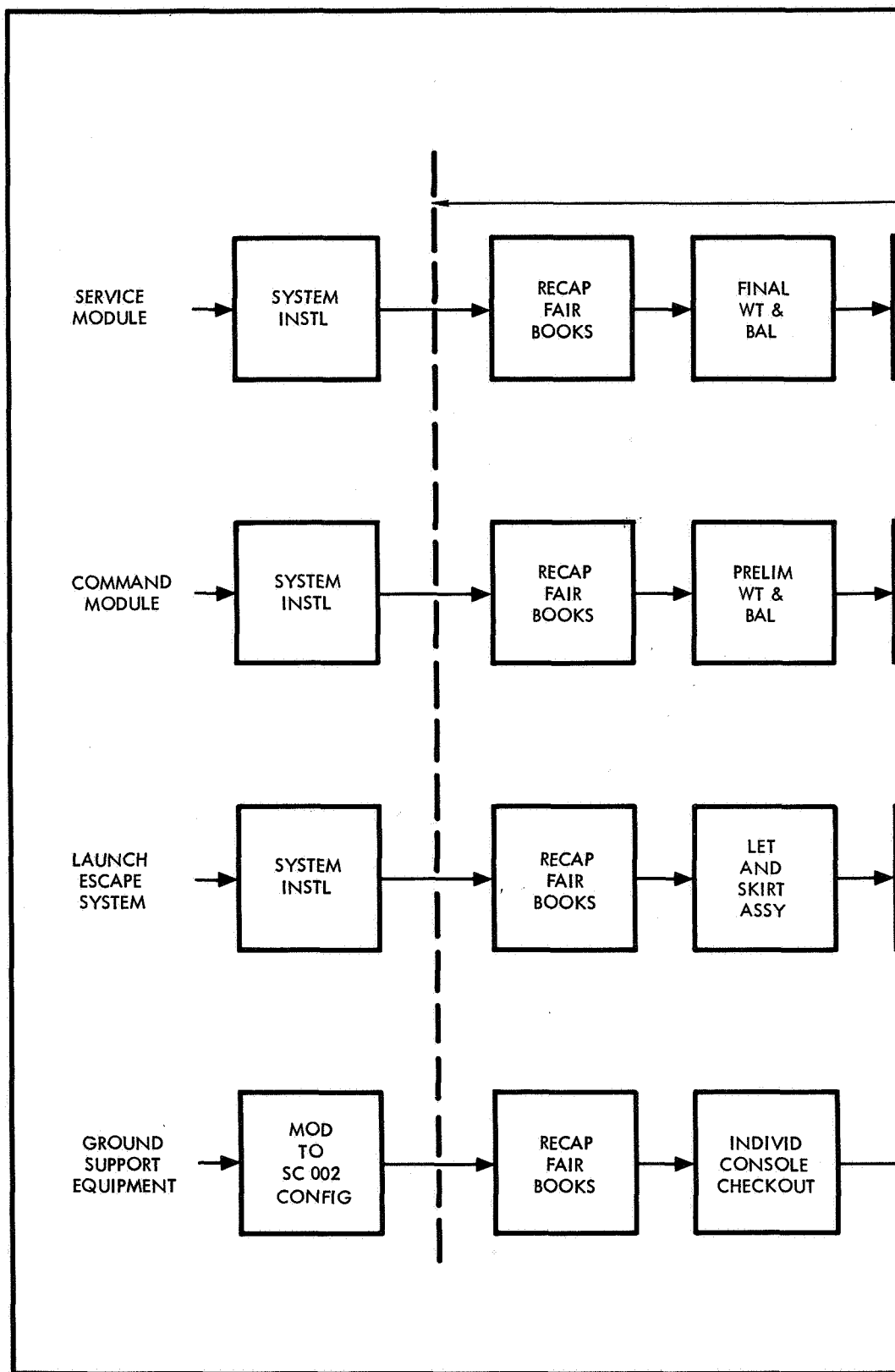


Figure 8-1. Downey Operations Schedule

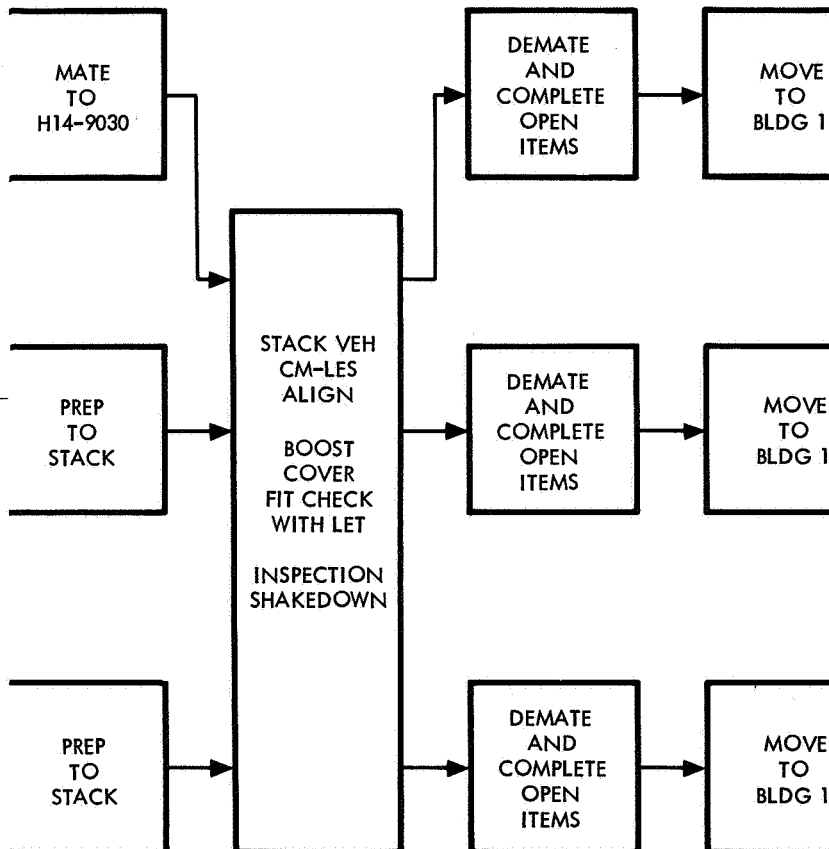
8-3,8-4

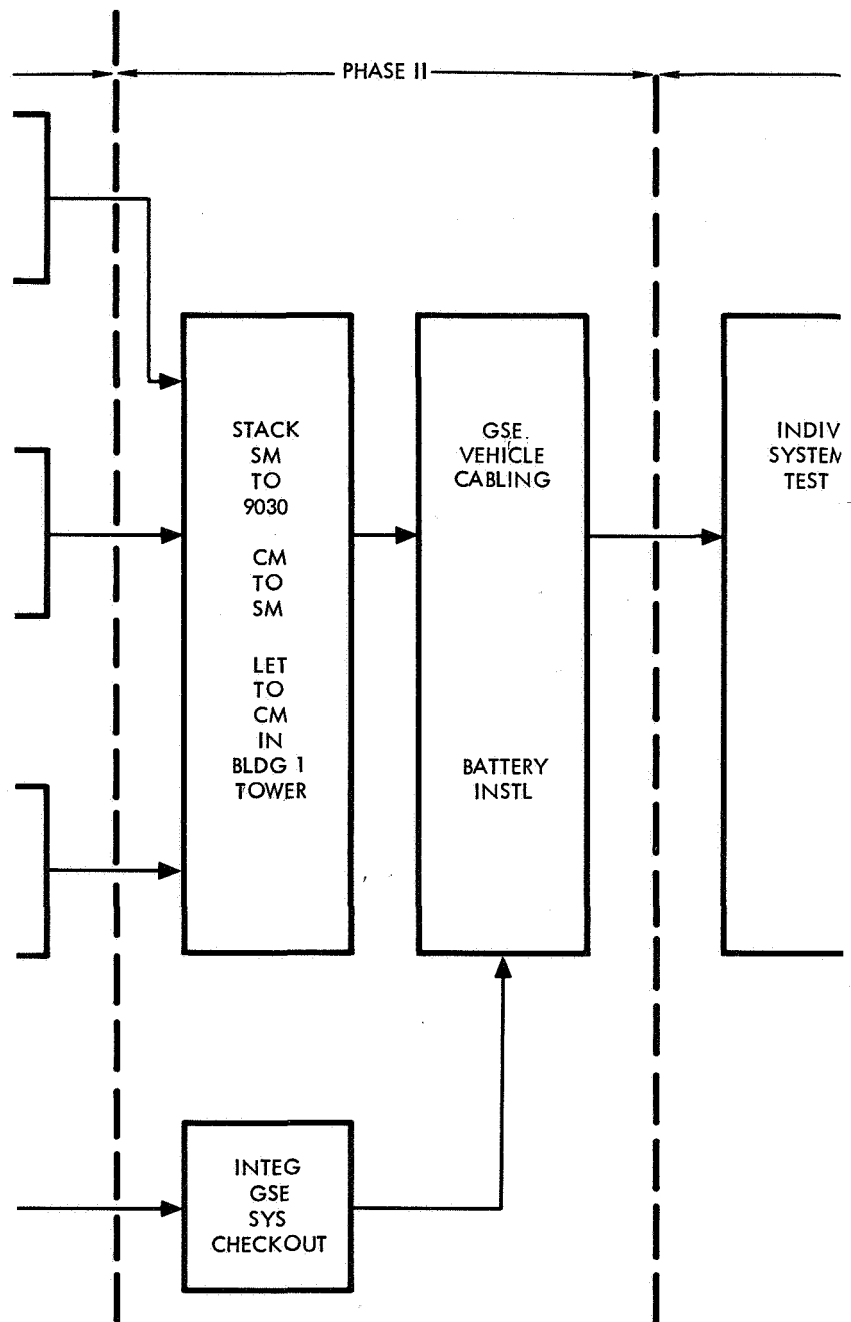
SID 64-2174



BUILDING 290

PHASE I





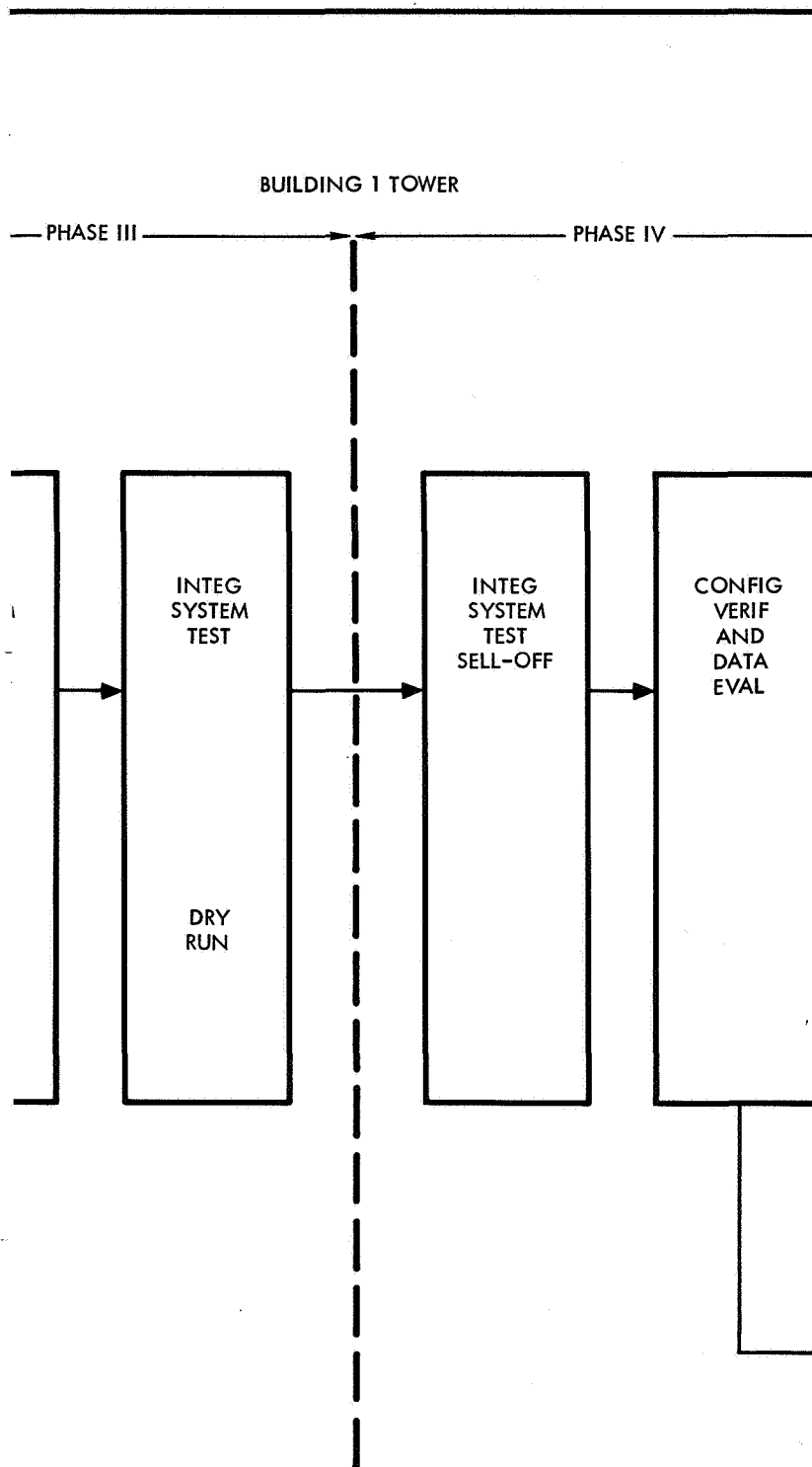
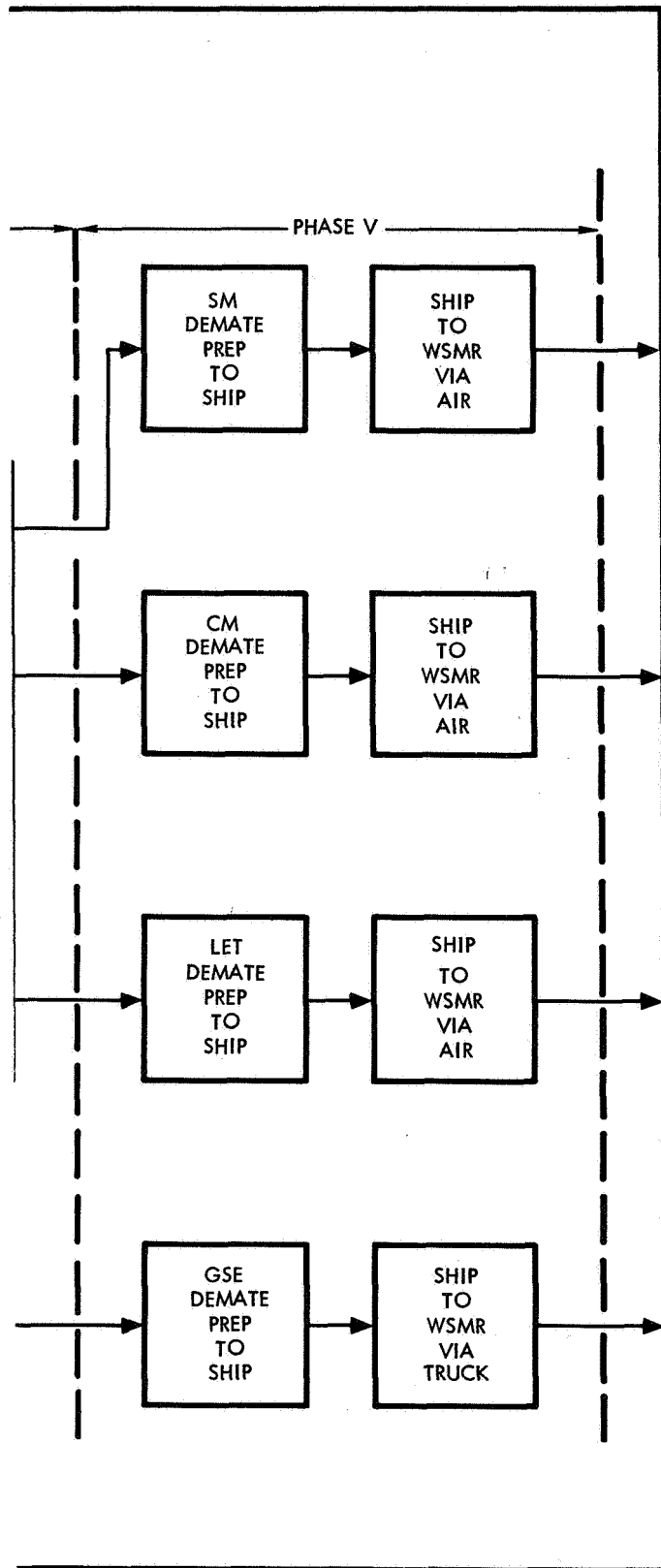


Figure 8-2. Prelaunch





Operational Flow, Downey



inputs to the GSE and verify outputs to the service module umbilical in order to establish interface compatibility. The inert motors will not be mated to the launch escape tower; however, the vehicle's electrical interfaces will be provided to the canard and Q-ball subsystems. These units and their instrumentation will be installed on a holding fixture located near the spacecraft stack in the tower. Upon completion of the GSE integrated systems test, the GSE-vehicle cabling hookup and battery installation will prepare the vehicle for system checkout.

#### 8.1.3 Third Phase

The third phase of test preparation will consist of spacecraft systems tests to verify individual system readiness. An electrical system test will verify the operation of the electrical power subsystem from ground power and from internal battery power. Power switching and activation of vehicle subsystems will also be verified. Electrical tests of the abort and earth landing sequences will be conducted to verify the proper occurrence of all of the abort sequences, including separation rocket-motor starts, forward compartment cover jettison, and parachute deployment. All testing will be accomplished with WSMR-type cabling and J-box equipment. Pyrotechnic simulators will be used to provide circuit loading and to give proper responses for launch escape subsystem and explosive devices testing. Both the proper phasing and timing of functions and redundant functions will be tested. Where interlocking circuits are provided to prevent a function from occurring out of sequence, the interlocking functions will be tested. During this testing, the explosive devices circuits will be monitored to determine whether or not interaction exists that might cause premature operation. Counters on the GSE will record the number of actuations of the firing units.

Instrumentation tests will include checkout and operation of the telemetry subsystems and checkout and operation of the onboard tape recorders. The vehicle antenna subsystems will be tested for proper output. A calibration of all end instruments will be accomplished. The C-band transponders will be functionally tested on bench maintenance equipment (BME). They will then be installed in the vehicle and checked as a vehicle subsystem.

#### 8.1.4 Fourth Phase

The fourth phase of test preparation will be the integrated systems compatibility test. The tests will be conducted with WSMR-type cables and J-boxes and will include sequential operation of all vehicle subsystems to verify subsystem compatibility.



### 8.1.5 Final Phase

The final phase will include preparation of the vehicle for shipment. The vehicle will be demated and those systems scheduled for separate shipment to WSMR will be removed and packaged. In general, the vehicle will be shipped with subsystems installed. A complete list of applicable documents and NAA drawing numbers will be found in the "Preparation for Delivery and Transportation of Apollo Spacecraft," MAD116-001 and MAD116-012.

All airborne items, i.e., launch escape subsystem, command module, service module, and service module extension, will be shipped by aircraft from Long Beach airport to Holloman Air Force Base. Transportation from S&ID, Downey, to Long Beach and from Holloman Air Force Base to the test site (WSMR) will be by truck. The spacecraft GSE will be shipped overland by truck to WSMR. Shipment plans and methods are subject to change; however, the most expeditious shipping plan compatible with the test schedule will be implemented. Day 1 of the field schedule indicates the arrival of the command module in the vehicle assembly building at WSMR. Other vehicle components and GSE will be received at WSMR before Day 1.

## 8.2 CHECKOUT OPERATION PLANS (DOWNEY)

The operational test procedures listed on Table 8-1 will be employed to establish the flight readiness of Spacecraft 002. Only those procedures denoted by the letter P in the Location column will be applicable for checkout at Downey.

## 8.3 VEHICLE PREPARATION (WSMR)

Preparation of Spacecraft 002 at WSMR will consist of two phases; spacecraft assembly building and spacecraft checkout and countdown at the launch site. The test sequence, from receipt of the vehicle at WSMR to launch countdown, is shown in Figures 8-3 and 8-4.

### 8.3.1 Vehicle Assembly Building Preparation

Upon arrival at WSMR, the command module and GSE will be delivered to the vehicle assembly building (VAB) for receiving inspection. The motors and ordnance will be delivered to the VAB from the ordnance storage area. Assembly of the launch escape subsystem, including the installation of all instrumentation and installation of parachutes and ordnance in the command module, will be performed in the VAB.

Final weight and center-of-gravity determination will be accomplished in the VAB. Weight and balance tests of the service module will not be



Table 8-1. Operational Test Procedures

Operational Checkout Procedure No. *	Location	Title
	P & A	Integrated System Tests
	A	Countdown Spacecraft 002 and Little Joe II
	A	Simulated Countdown, Spacecraft 002 and Little Joe II
	A	Pyro Receiving Inspection, Storage & Preinstallation Checkout
	A	Electrical System Checkout
	P	Electrical Verification & Functional Power Checks
	P & A	LES & ELS Sequencer Verification Checks
	P & A	BME Checks ELS Sequencer and Baroswitch
	P & A	Bonding Verification of Spacecraft and GSE
	P & A	Battery Service
	P & A	LES Building Up
	A	CM & LES Vertical Move
	A	CM-SM-LET Receiving Inspection
	P	SM Weight & Balance
	A	SM Mate and Demate to Little Joe II
	P & A	Test Vehicle to SM Mate & Demate & Alignment Checks
	A	Handling and Transportation of the Test Vehicle
	P	Forward Compartment Cover Installation, Removal
	P & A	Forward Compartment Equipment Installation, Removal
	P & A	CM to LES Mate and Demate
	P & A	CM Weight Fixture Installation and Alignment
	P & A	CM Weight and Balance (Horizontal)
	P & A	CM Weight and Balance (Vertical)
	A	LES Weight Fixture Installation and Alignment
	A	LES Weight and CG Test (Horizontal)
	P & A	Thrust Vector Alignment Gross Weight & CG Determination
	A	Preinstallation Leak Check of LES Motors
	P & A	CM to SM Mate and Demate
	A	Functional Verification of Q-Ball
	A	Functional Verification of Camera System
	P & A	Functional Verification of Transponder
	P & A	Functional Verification of Antenna System
	A	Functional Verification of Telemetry System
	P & A	Functional Verification of Instrumentation System
	P & A	BME R&D Tape Recorder
	P	Functional Verification of Telemetry & Instrumentation System
	P & A	Tape Recorder Loading
	P	Functional Verification of Telemetry Ground Station
	P & A	Test Configuration Checklist
<p>*The OCP's listed will in most cases be a rewrite of existing OCP's. The OCP numbers have not been assigned. This information will be supplied when it becomes available.</p> <p>P Downey A Field</p>		





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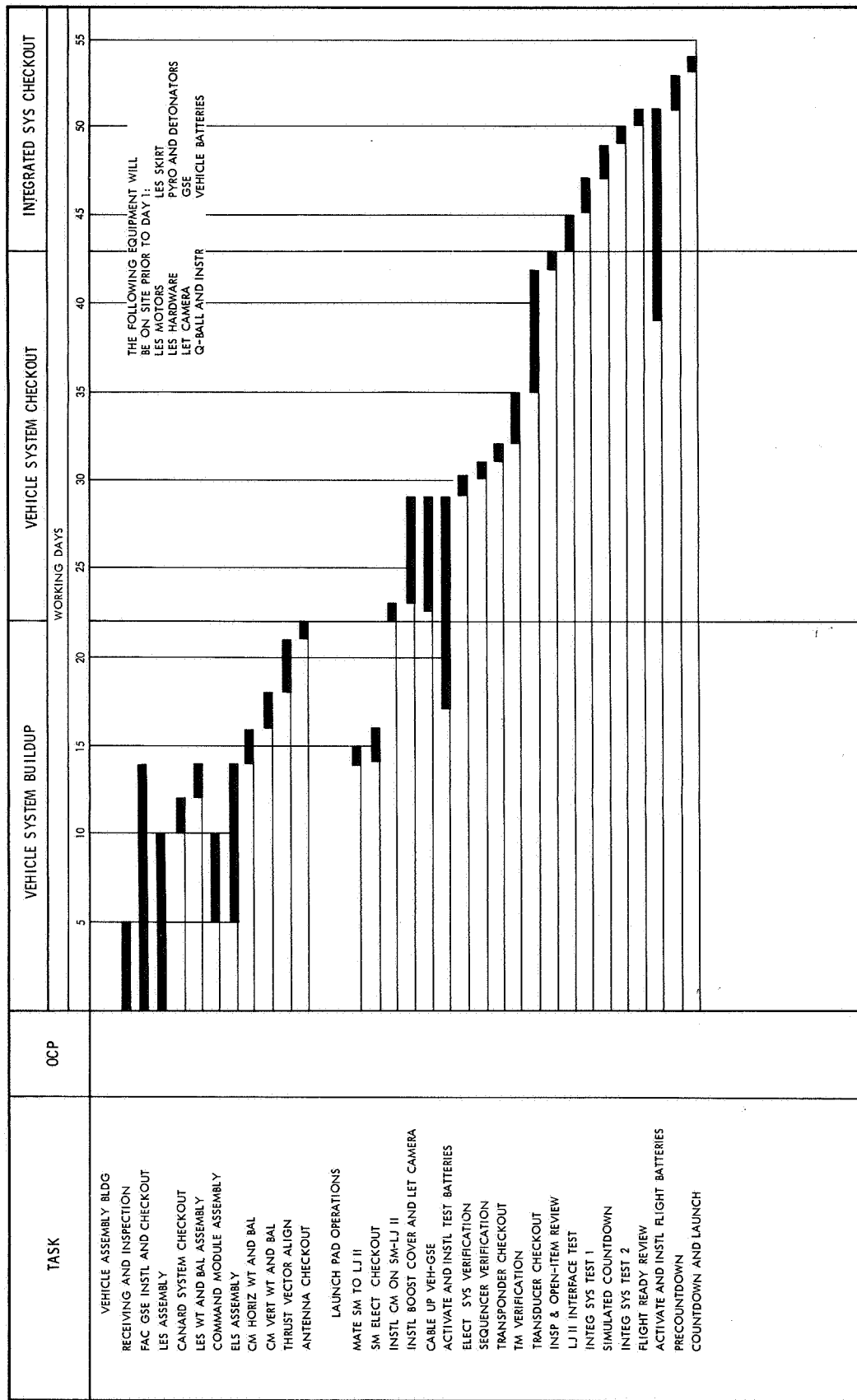
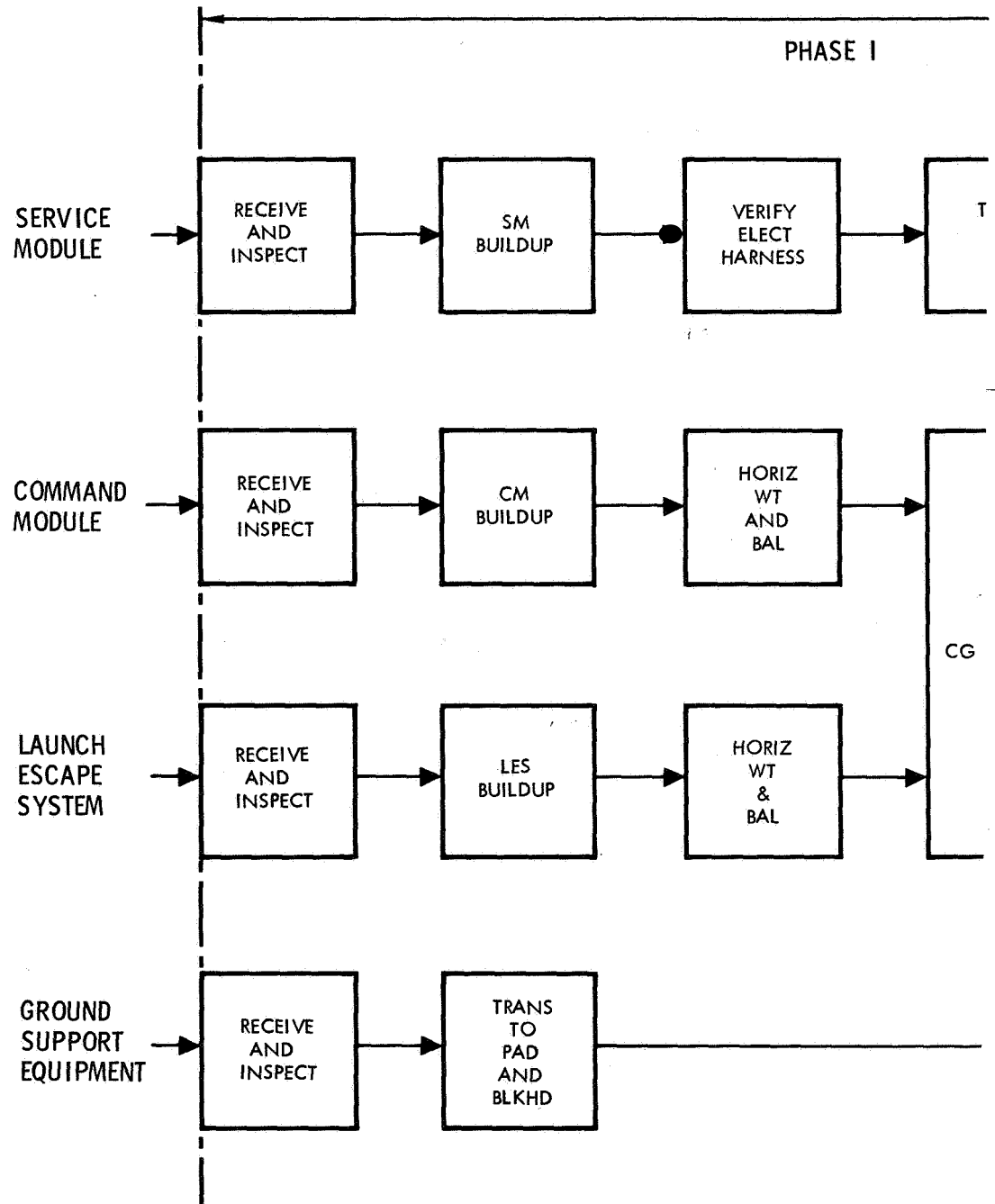


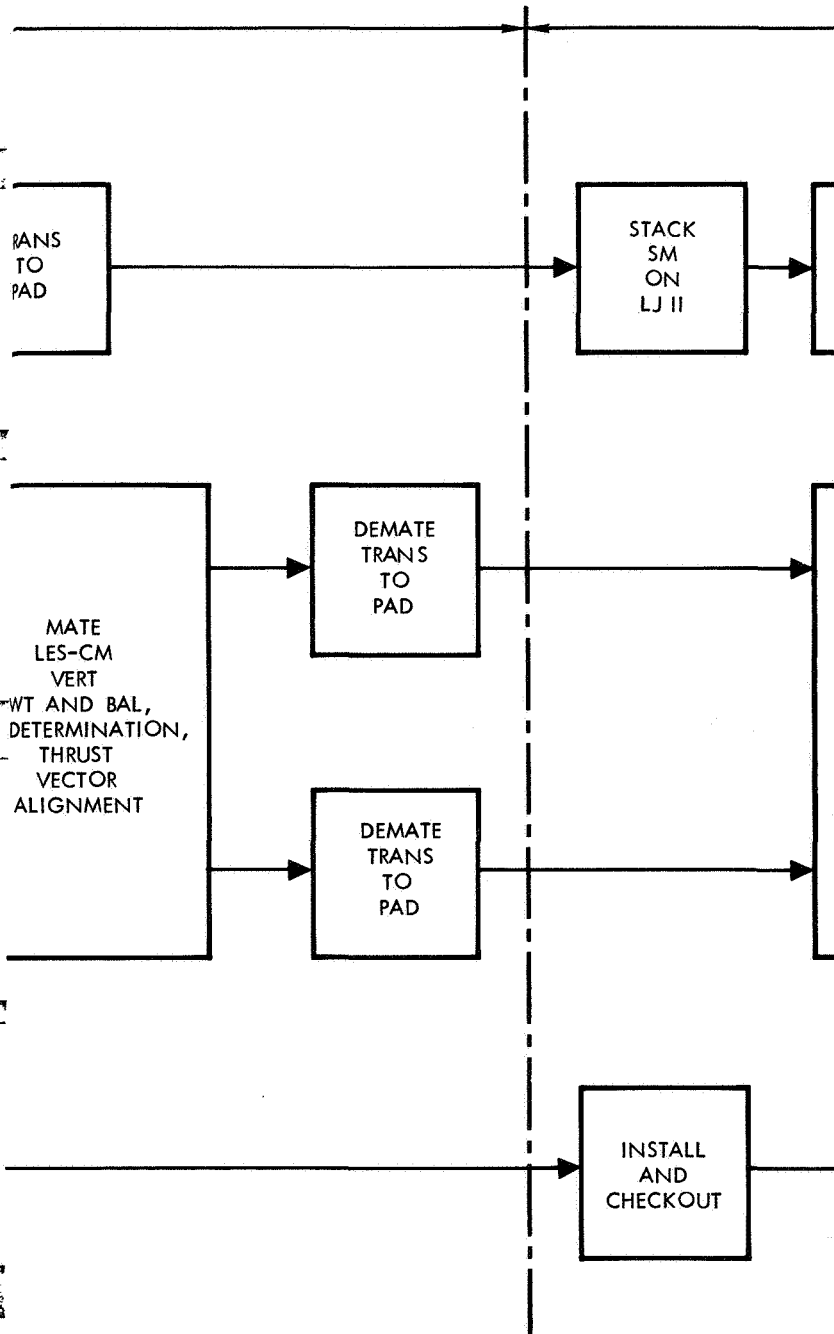
Figure 8-3. WSMR Operations Schedule

8-11, 8-12

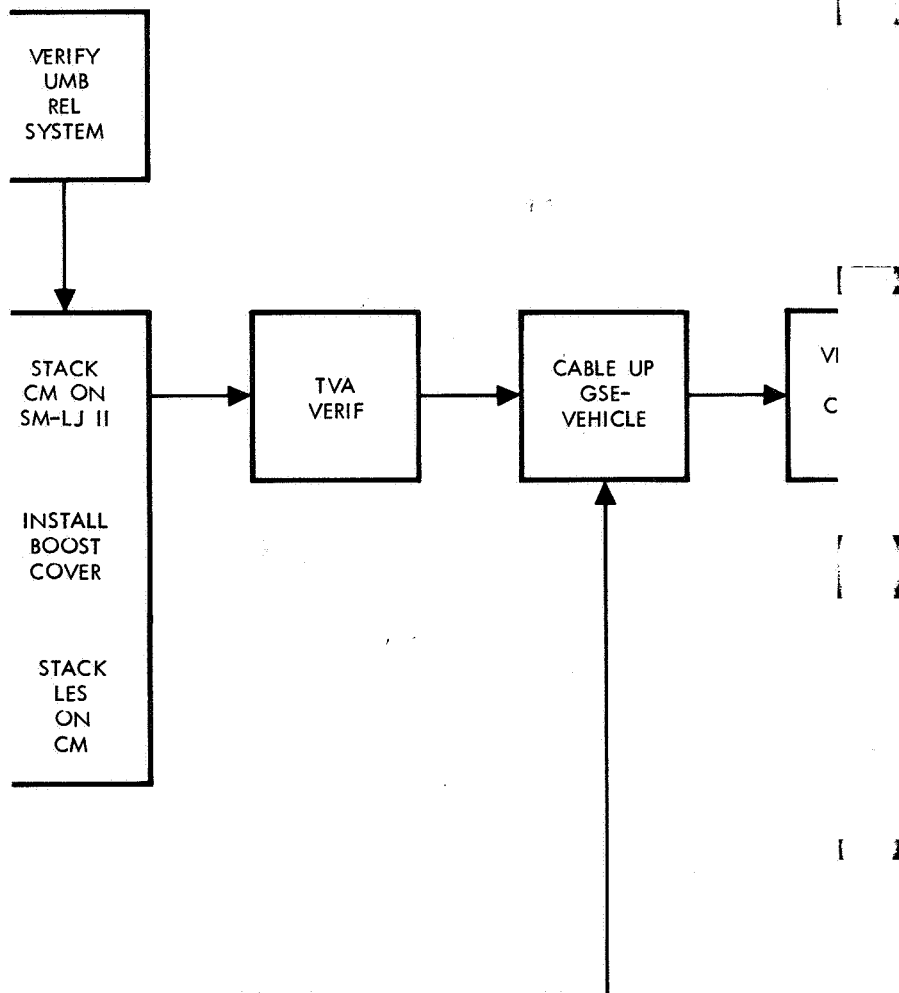
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# VEHICLE ASSEMBLY BUILDIN









LAUNCH COMPLEX 36

PHASE II

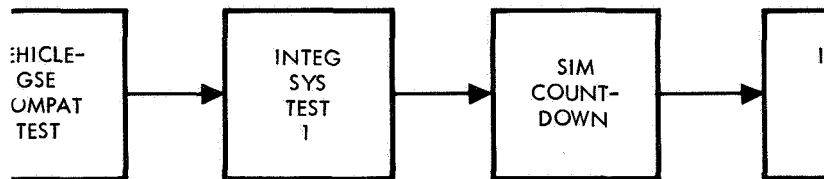
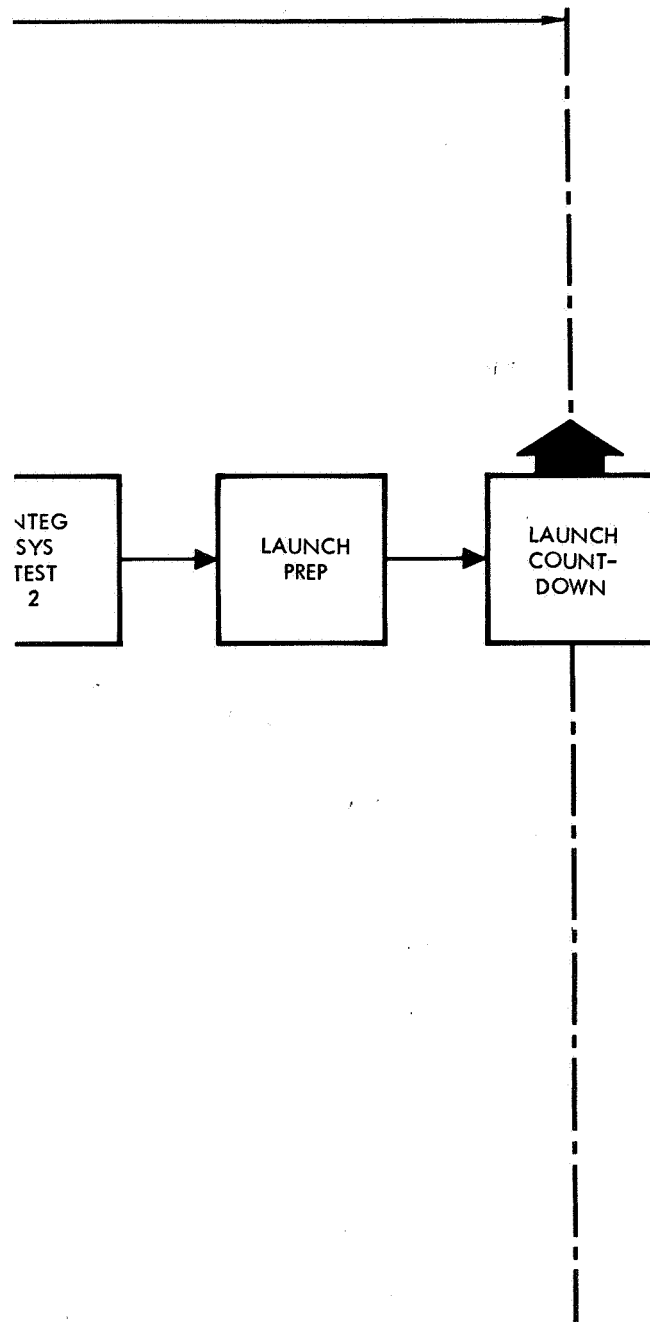


Figure 8-4. Prelaun



ch Operational Flow, WSMR



performed unless extensive configuration changes are made after Downey weight and balance. The launch escape subsystem will be weighed in the horizontal position and the command module will be weighed in the horizontal and vertical position. A vertical weight and balance will then be performed on the launch escape vehicle. Based on the center of gravity established from these final weight and balance tests, the thrust vector alignment index of the launch escape motor will be defined. Optical means will be used to project this thrust vector to a point on the command module, so that this alignment can be reaffirmed after stacking.

The service module will be transported to the launch pad for mating upon completion of Little Joe II buildup. After alignment testing, the assembled command module and launch escape subsystem will be demated and transported to the launch pad for mating operations.

#### 8.3.2 Vehicle Launch Site Preparation

Upon delivery of the launch escape subsystem and the command module to the launch pad, the test vehicle will be mated to the service module and Little Joe II combination. A thrust vector verification test will be performed, and the boost protective cover will be installed. Vehicle-GSE cabling will be accomplished. Calibration of the instrumentation subsystem will be reverified to be at launch level through the pad facilities and the vehicle telemetry subsystem. Subsystem safety verification tests will be conducted to ensure checkout equipment compatibility and to verify subsystem operations. An integrated systems test will then be conducted to verify test vehicle readiness. This test will be followed by a simulated countdown and a final integrated systems test. A flight readiness review meeting is the final declaration that the vehicle is ready for launch countdown. The launch countdown is the concluding activity that verifies that the vehicle has the ability to operate within its performance specification during flight.

### 8.4 FIELD OPERATION PLANS (WSMR)

Field operation for flight readiness includes procedural checkout and launch day activity.

#### 8.4.1 Procedural Checkout

The operational test procedures of Table 8-1 denoted by the letter A in the Location column will be applicable for checkout at WSMR.

#### 8.4.2 Launch Day Activity

On launch day, Spacecraft 002 launch countdown will be conducted, including preparation of all subsystems for launch. Range support agencies



will be on station to support the prelaunch activities and the flight and post-launch activities. At a predetermined time before launch, the launch pad area will be cleared of all personnel and equipment. The final phase of checkout and launch will be conducted from the blockhouse. The precount and countdown to launch will proceed as shown in Figures 8-5 and 8-6.

The countdown embodies the detailed checkout and verification of flight readiness of the spacecraft and the boost vehicle. The details are accounted for separately and checked off step by step (1 through 30) in the task groups. Not all of these tasks are S&ID responsibility; however, they can be performed concurrently with tasks that are the responsibility of General Dynamics, Convair (GDC).

The combined efforts are coordinated through the NASA test director, who has the responsibility to initiate the start of the countdown, to conduct evaluation activities, and to control the countdown from initiation to the announcement of "clear to launch."

#### 8.4.2.1 Range Support and Requirements

Range support requirements will be identified and detailed by the NASA document entitled "Program Requirements Document" (PRD) and this S&ID Vehicle Test Plan document (SID 64-2174, Appendix B). The areas in which support is required are as follows:

- Photographic coverage
- Optical and camera tracking
- Telemetry recording
- Range and pad safety
- Meteorological support
- Radar tracking
- Vehicle recovery

Before launch countdown, the NASA Test Director will coordinate the testing sequence with the range support groups to ensure compatibility for tracking and data acquisition readiness in the vehicle tracking station and the telemetry station. It is also the responsibility of NASA to deliver tracking, telemetry, and photographic information to the S&ID Data Engineering group within 12 hours after launch. This information is required to expedite the completion of the postlaunch reports.

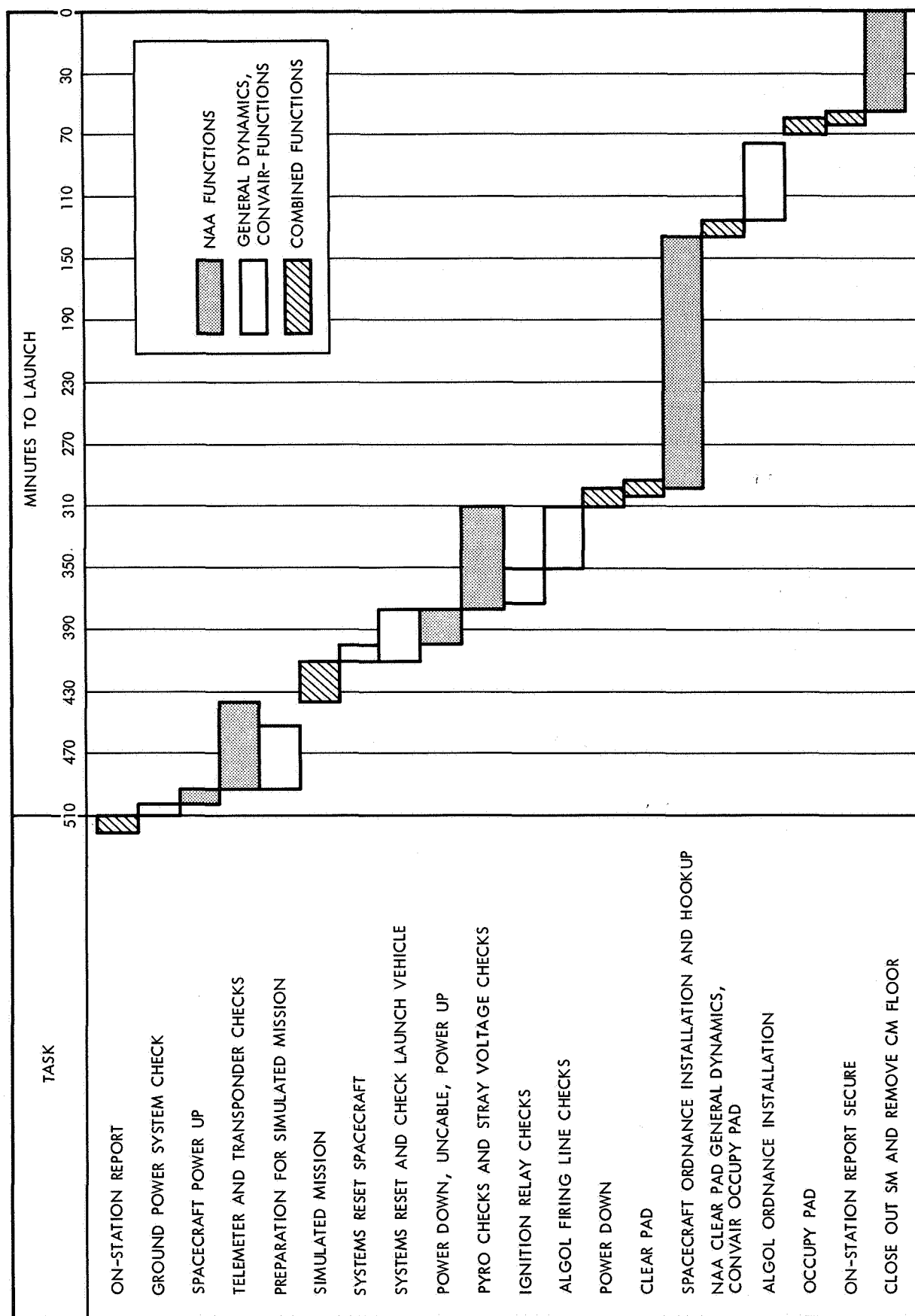


Figure 8-5. Precountdown Task Sequence

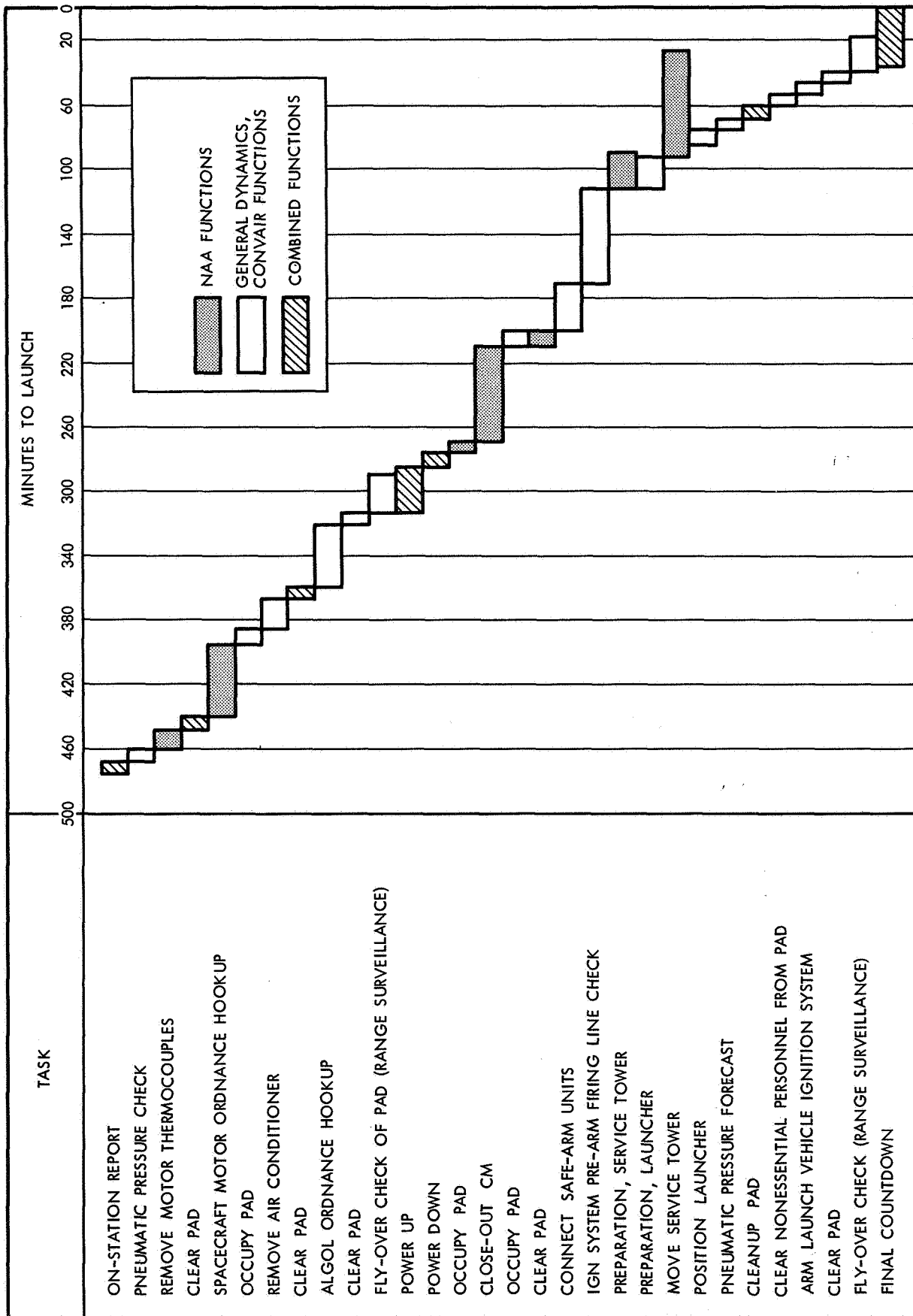


Figure 8-6. Countdown Task Sequence

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## 9.0 FLIGHT TRAJECTORY REQUIREMENTS

### 9.1 TRAJECTORY DESIGN GROUND RULES

The power-on tumbling boundary is defined by abort initiation conditions for tumbling aborts where the combined aerodynamic and launch escape motor exhaust pressures imposed on the command module surface approach the design limit range of 9 to 11 psid across the command module outer structure. The recommended abort initiation conditions superimposed on the Saturn boost trajectory envelope is presented in Figure 9-1. The abort initiation conditions, with the recommended test point located, is presented in Figure 9-2.

### 9.2 ABORT ENVELOPE

Power-on tumbling boundary abort mission A-004 can be successfully accomplished, not only from precisely the abort point but from any point within the envelope of allowable dispersion. Dispersion can result from tolerances in the Little Joe II control subsystem and/or in the abort initiation timer, from performance variations of the Algol rockets, from pressure, density, drag, from wind conditions, or from any combination of these factors. The performance variation of a typical Algol rocket is shown in Figure 9-3. Rocket-to-rocket variation can result from mass flow coefficient, burning rate constant, and throat area. The acceptable dispersion values and envelope for this mission have not been established at this date.

### 9.3 CONTINGENCIES

The design or primary mode of abort initiation will be radio command to the Little Joe II booster for accomplishment of the pitch maneuver. A real-time display of vehicle altitude and velocity will be monitored for optimum moment radio command transmission for initiation of the pitch maneuver to account for vehicle dispersions. The Little Joe II abort timer will be actuated upon start of the pitch maneuver so that abort initiation will occur 3.5 seconds after pitch initiation and approximately 76.0 seconds after vehicle lift-off.

The backup abort timer aboard the Little Joe II will generate the command for the pitch maneuver and actuate the abort delay timer if the





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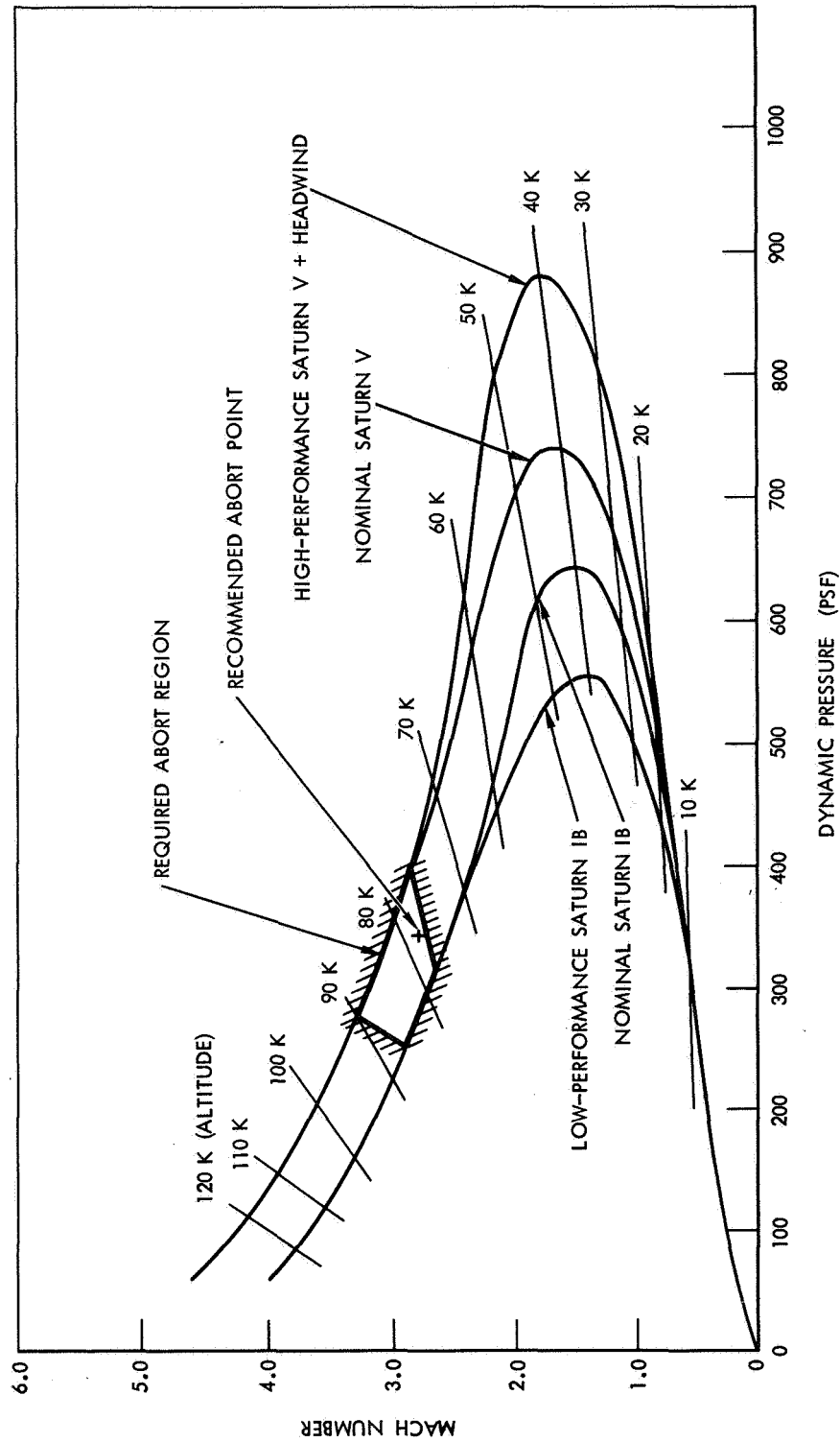


Figure 9-1. Saturn Boost Preliminary Trajectory Envelope

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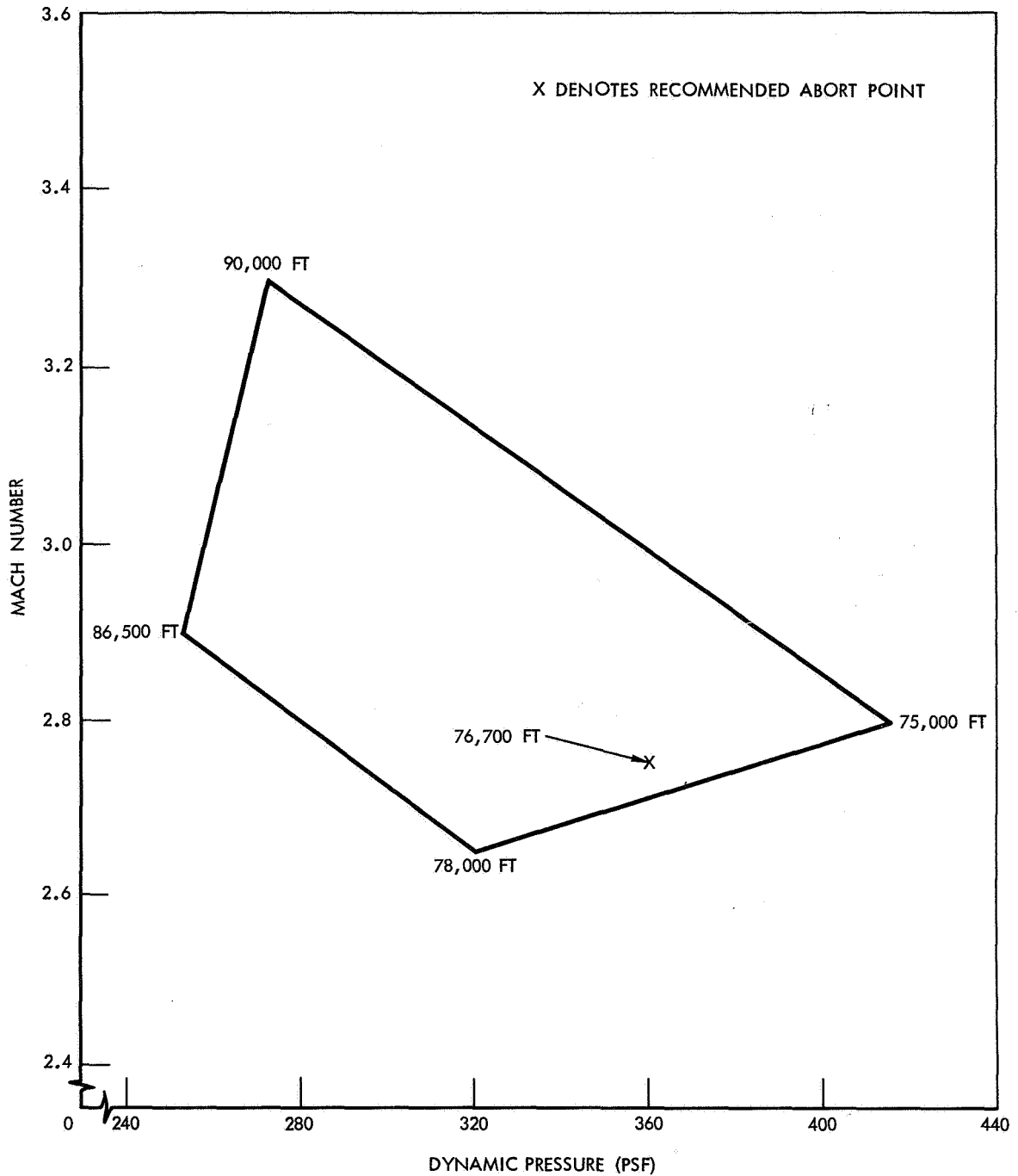
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Figure 9-2. Power-ON Tumbling Boundary Abort Preliminary Envelope

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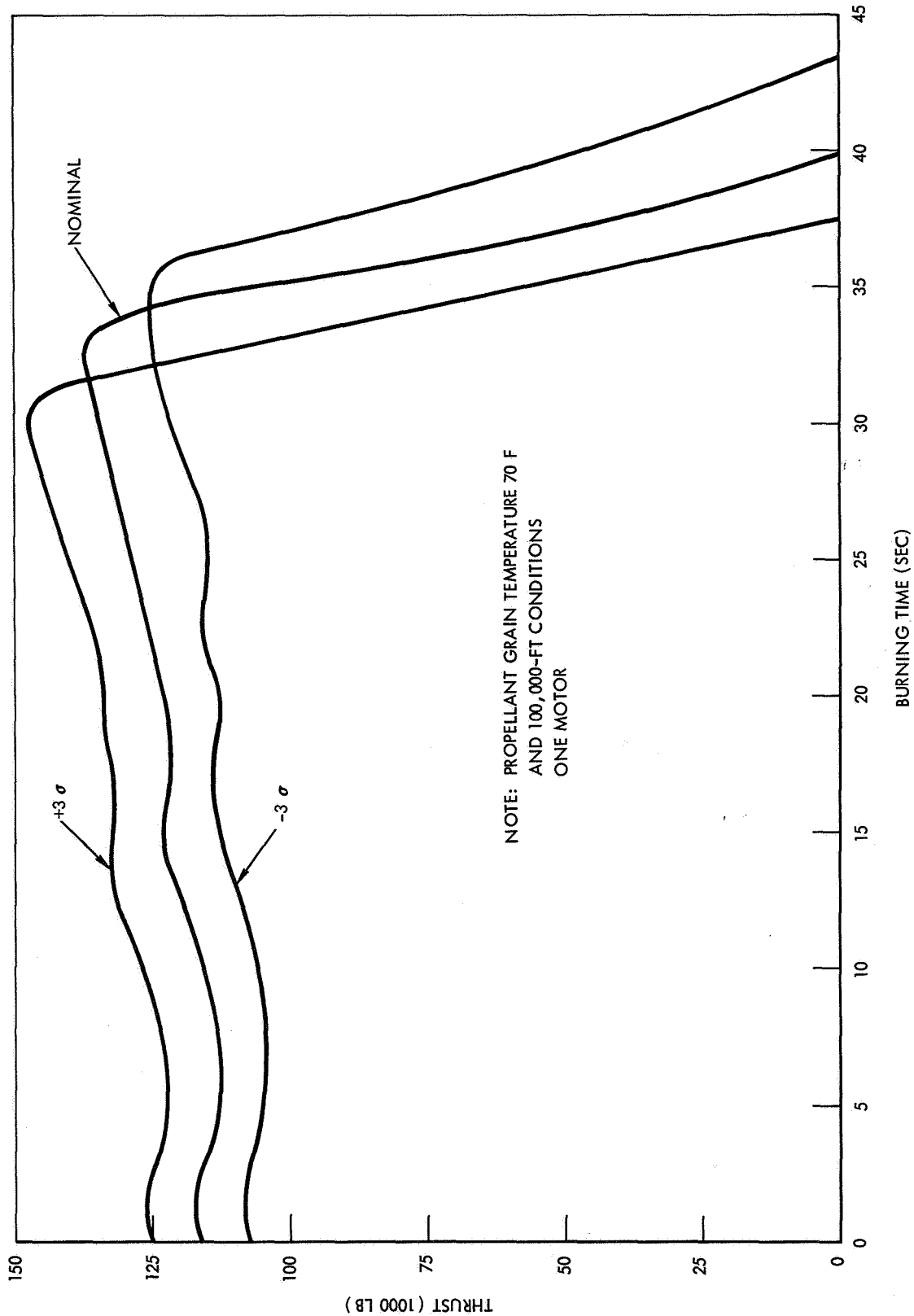


Figure 9-3. Typical Algol Thrust Versus Time (Preliminary)

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radio signal is withheld or fails to reach the vehicle. The backup timer will become active at a predetermined time after lift-off. This time has not yet been established.

#### 9.4 GROUND REAL-TIME DISPLAY

The real-time displays at the ground station will be those necessary for abort initiation and for range safety. Plotting boards and a digital display panel will be employed at WSMR. The following information will be available on the plotting boards:

1. Vehicle altitude versus velocity
2. Cross-range distance and altitude versus down-range distance (two plots)

#### 9.5 RANGE COVERAGE PLOTS

Spacecraft 002 will be launched from WSMR launch complex 36. A plot of the range, including the tracking stations and a ground trace of the flight, is contained in Figures 9-4 and 9-5.

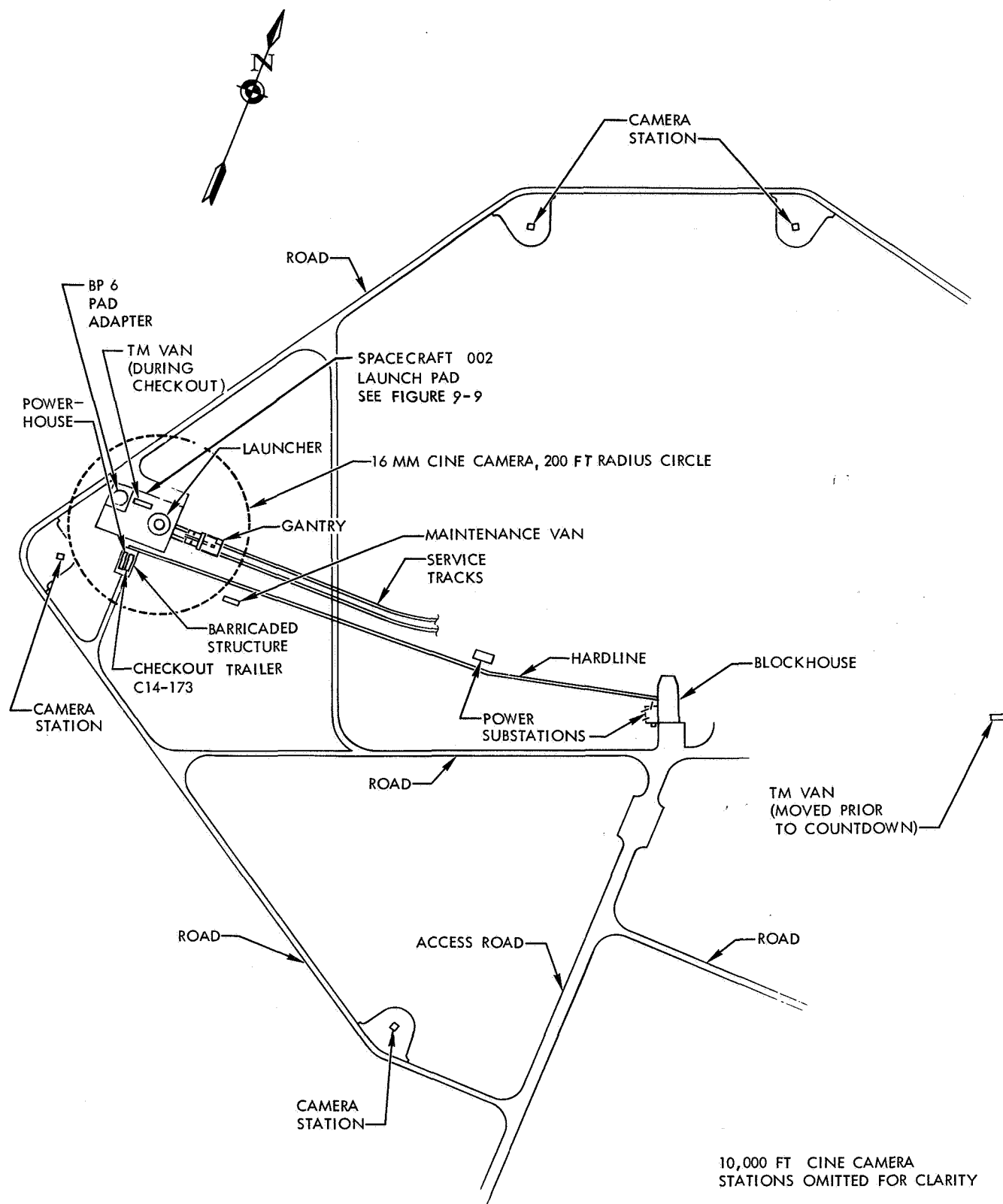


Figure 9-4. Launch Complex 36, WSMR

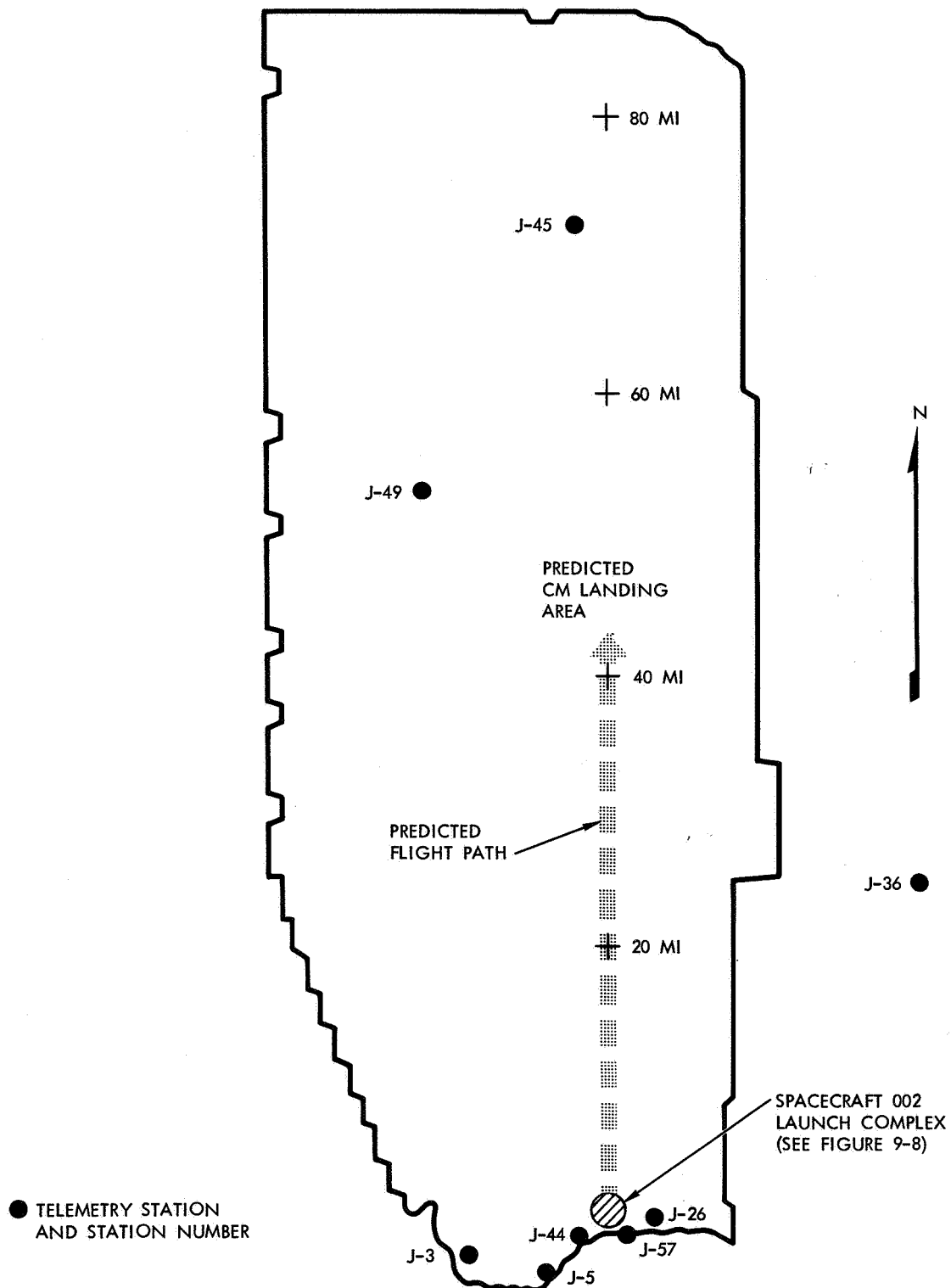


Figure 9-5. Launch Site, WSMR

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## 10.0 FLIGHT PLANS

### 10.1 MISSION DESCRIPTION

Apollo power-on tumbling boundary abort mission A-004 is summarized Figure C-2 of Appendix C. This section describes the mission in detail. Mission A-004 will comprise the following phases of flight:

- Little Joe II booster launch
- Little Joe II pitch maneuver
- Launch escape vehicle tumbling abort
- Launch escape vehicle orientation and stabilization
- Launch escape subsystem jettison and forward heat shield separation
- Parachute deployment and command module deceleration
- Command module touchdown and recovery

The Launch Escape Vehicle - Little Joe II stack resting at 84 degrees from downrange horizontal will be launched from the White Sands missile range (WSMR) at 4036 feet elevation above mean sea-level. Two of the four Algol engines contained in the Little Joe II will be ignited simultaneously for lift-off and initial boost by signal from the blockhouse. The third and fourth Algol engines will be ignited simultaneously 36.5 seconds after lift-off, prior to burnout of the first two Algols, which occurs at 39.5 seconds after lift-off. The booster will maintain the programmed trajectory to the abort point. As the booster reaches approximately 71,000 feet at 72.5 seconds after lift-off, a radio command will activate the booster control subsystem to affect a pitch maneuver to a negative angle-of-attack. The booster-contained backup timer will be set for a predetermined time interval after lift-off to initiate the pitch maneuver if the radio command fails to affect the maneuver. Initiation of the pitch maneuver, either by radio command or backup timer, activates the booster-contained delay timer for abort initiation 3.5 seconds following pitch initiation.

Abort will occur at around 75,000 feet altitude, at 76.0 seconds after lift-off. Vehicle speed will be approximately Mach 2.75, and the dynamic pressure will be approximately 360 pounds per square foot at the abort point. Burnout of the third and fourth Algol engines (booster second stage) will be at the same approximate time as abort initiation. Escape will be accomplished by firing the launch escape motor, propelling the launch escape vehicle (LEV) on an upward escape course beyond the booster. Because of

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LEV instability brought about by a change in the angle-of-attack (Little Joe II pitch maneuver) and a shift in the center of aerodynamic pressure on the LEV through escape motor exhaust plume envelopment, the vehicle will be expected to tumble end-over-end through and after escape motor combustion. Propellant burn time of the launch escape motor will be approximately seven seconds. The LEV will continue its tumbling course on a coast-up after escape motor burnout to an apogee of approximately 120,000 feet.

At 11 seconds after abort initiation on coast-up, the canard surfaces will be deployed for vehicle stabilization and orientation. The vehicle will continue to tumble, however, because the canards will have very little or no aerodynamic effect in the low dynamic pressure region of the trajectory. As the vehicle descends into denser atmosphere around 90,000 feet and below, tumbling will be arrested and the vehicle eventually stabilized and oriented with the aft (main) heat shield in the direction of descent.

As the LEV drops to the 24,000-foot level, baroswitch actuation detonates and fractures the dual-mode bolts, disjoining the escape tower from the command module. The baroswitch action will also result in jettison of the LES and attached boost protective cover as one unit by LES jettison motor firing. The command module forward heat shield will be ejected by thruster firing 400 milliseconds after LES and BPC jettison. The mission sequencer will be mechanized to transmit the thruster firing signal 0.4 seconds after transmission of the jettison motor firing signal in order to afford heat shield protection of earth landing equipment from the BPC as it sweeps and brushes by during jettison. Two seconds after LES jettison, the two drogue parachutes will be deployed by mortars in a 40 percent reefed configuration to minimize opening shock loads. Disreefing to the nominal 13.7-foot diameter will occur 8 seconds after line stretch. As the command module drops through 11,000 feet, the drogues will be released by actuation of a second baroswitch. Simultaneously, the three pilot parachutes will be deployed by mortars, each pilot extracting one main parachute in a 9 percent reefed configuration. The main chutes will be disreefed to the nominal diameter of 85.5 feet 8 seconds after line stretch. Command module touchdown will occur approximately 582.0 seconds after lift-off at approximately 42 miles from the launch point.

## 10.2 MISSION TRAJECTORIES

Mission preliminary trajectory data from test vehicle lift-off to command module touchdown for power-on tumbling boundary abort mission A-004 are contained in Figures 10-1 through 10-4. Figure 10-1 is the mission flight profile, and Figures 10-2 through 10-4 are the aerodynamic parameter time history traces.





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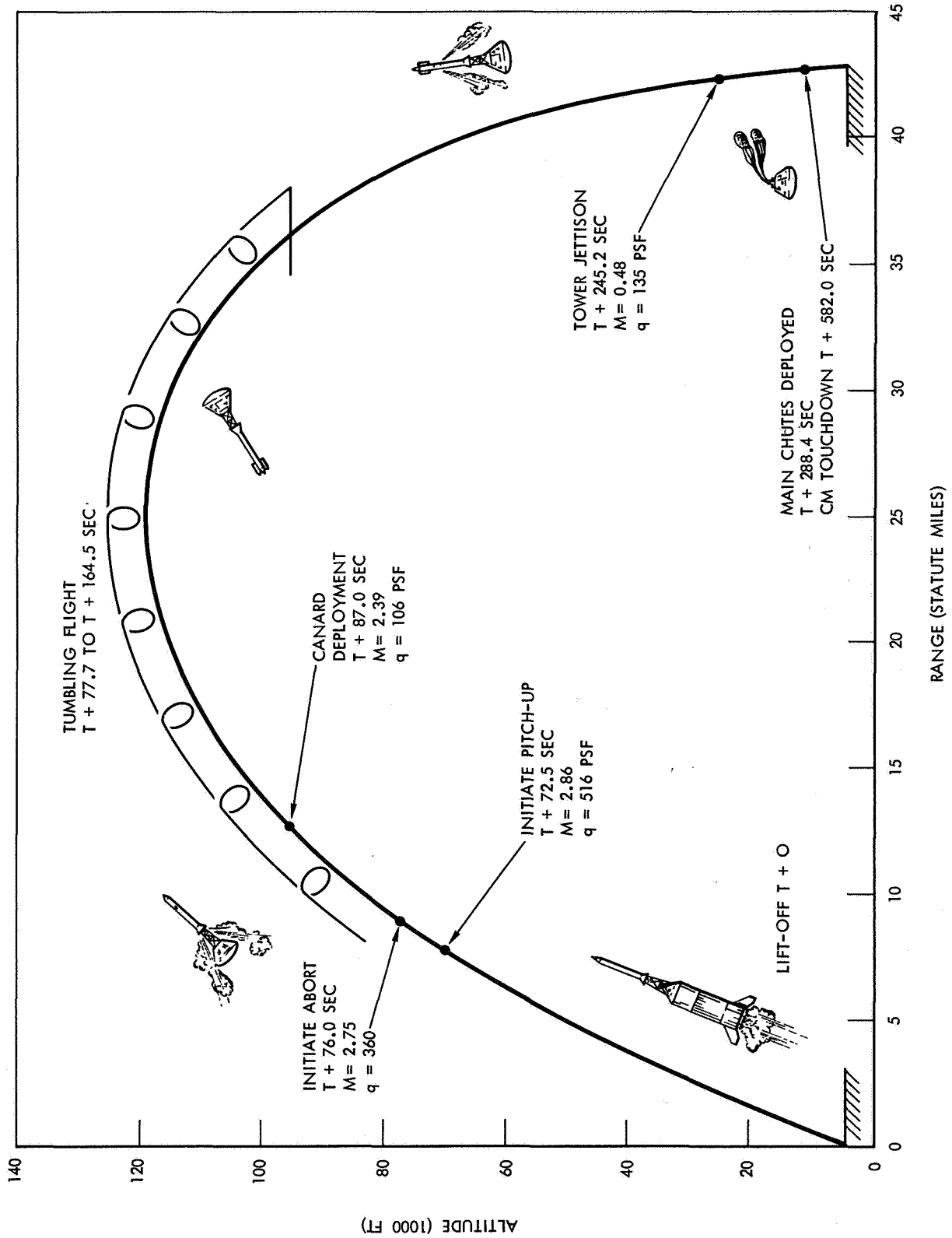


Figure 10-1. Preliminary Mission Profile



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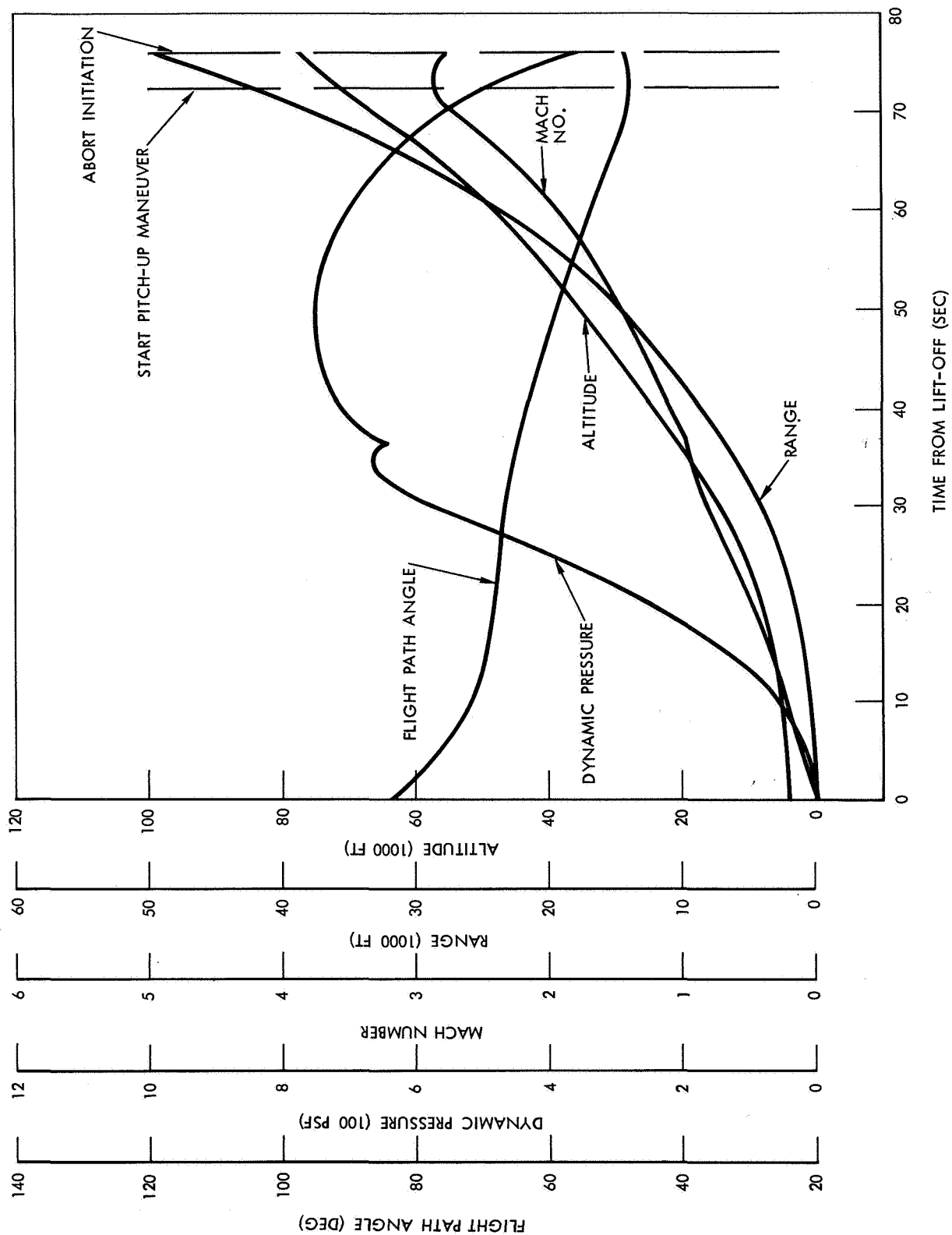


Figure 10-2. LEV Nominal Boost Trajectory Parameters (Preliminary)

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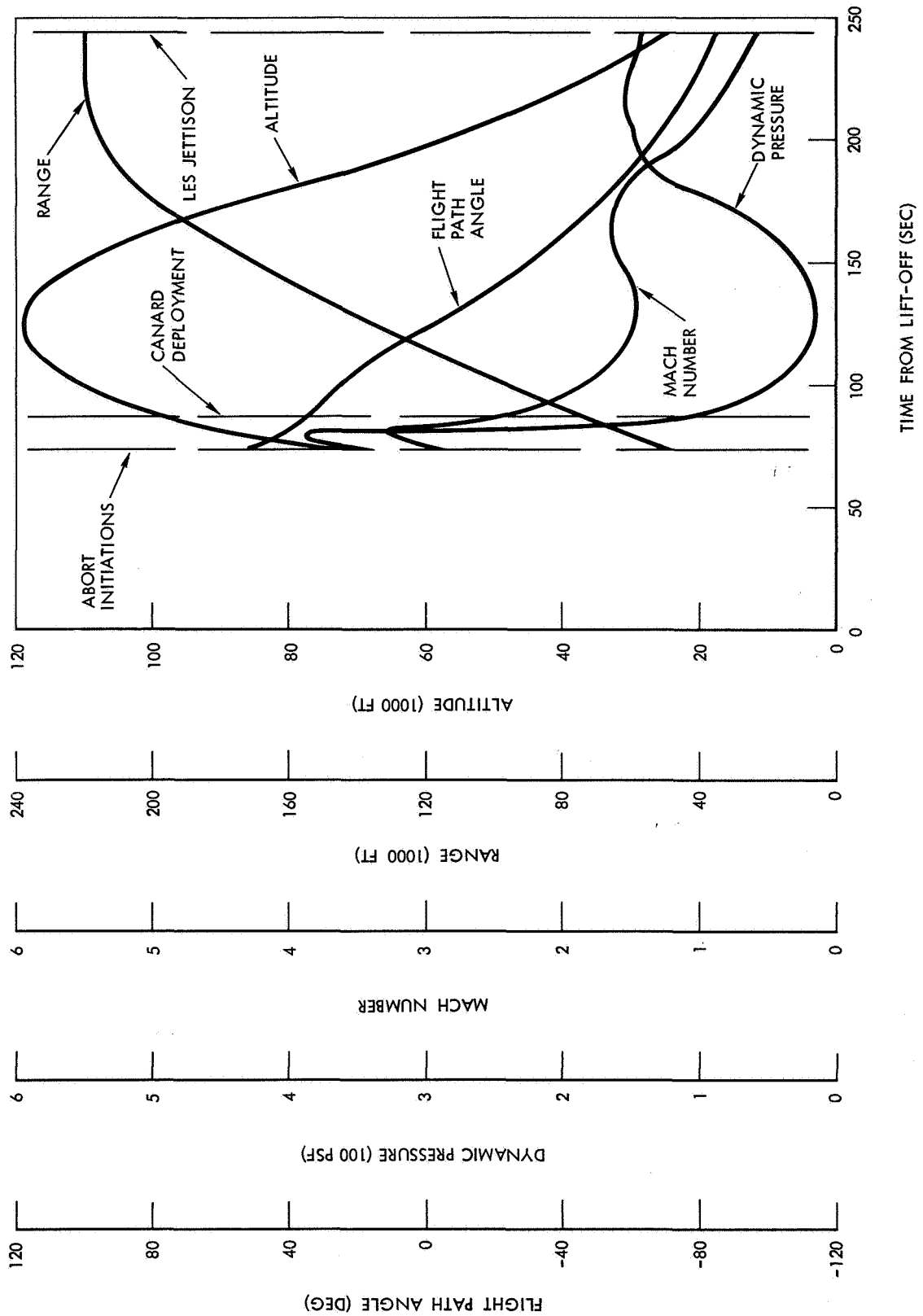


Figure 10-3. LEV Nominal Abort Trajectory Parameters (Preliminary)

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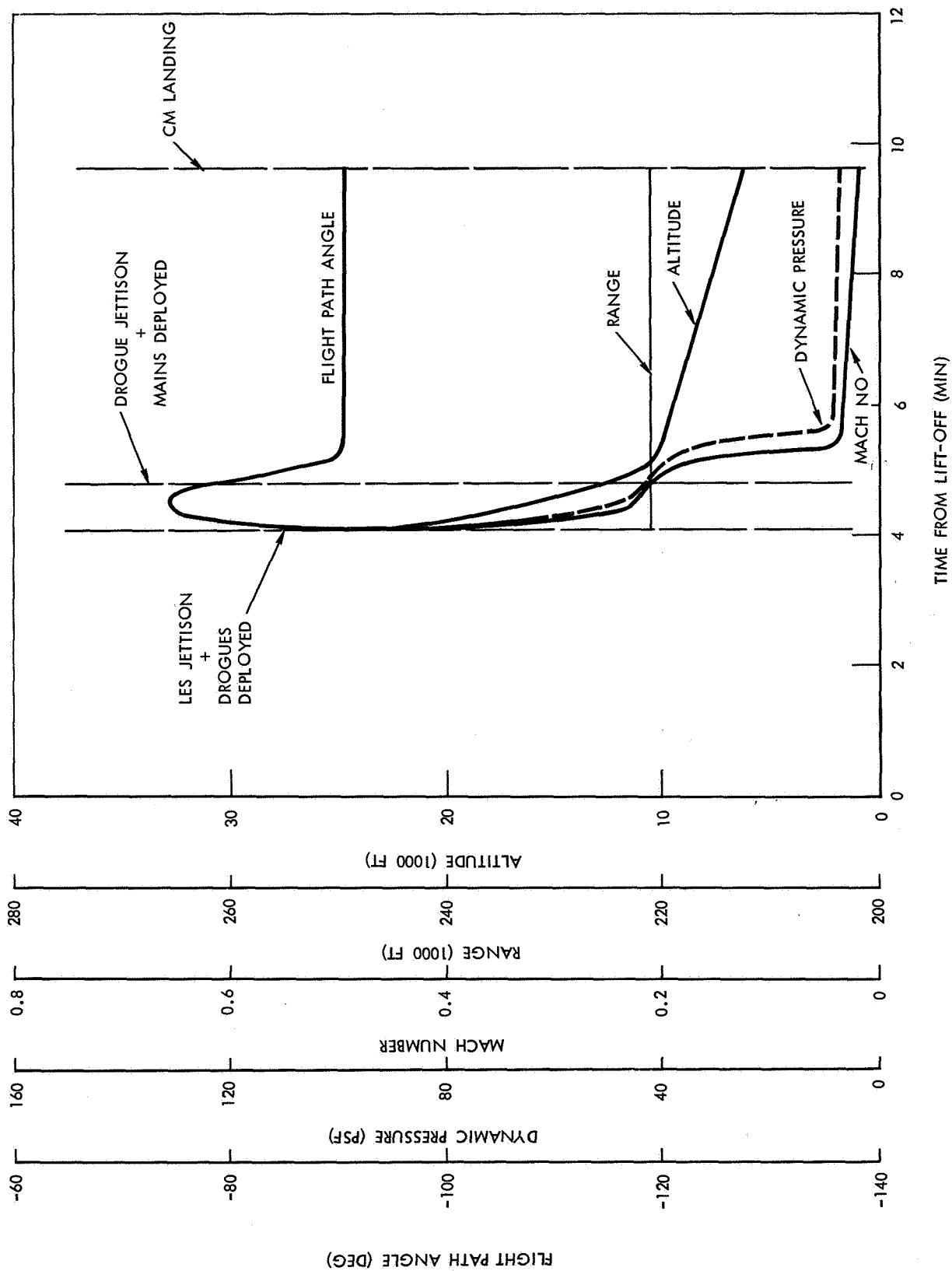


Figure 10-4. LEV Nominal Recovery Trajectory Parameters (Preliminary)

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### 10.3 SEQUENCE OF EVENTS

The nominal sequence of mission events with associated flight parameter values is contained in Table 10-1. The mission sequencer subsystem block diagram of Figure 4-7 presents the flow path of each event.

### 10.4 OPERATIONAL TIMELINES

The operational timeline history for this mission is presented in Figure 10-5. Included are timelines for all mission events and for vehicle-contained information transmission and recording equipment duty periods.

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Table 10-1. Nominal Sequence of Mission Events (Preliminary)

Time (sec)	Event	Mach No.	Dynamic Pressure (psf)	Altitude (ft)	Range (ft)	Flight Path Angle (deg)
T -	Little Joe II ignition (2 Alcol engines)	0	0	4,036	0	84.0
T + 0	Test vehicle lift-off		0	4,036	0	84.0
T + 36.5	Ignition of third and fourth Alcols (Little Joe II)	1.0	630.0	20,000	6,500	64.0
T + 72.5	Test vehicle pitch-up initiation	2.85	500.0	71,000	42,000	48.5
T + 76.0	Abort initiation	2.75	360.0	76,700	48,834	48.2
	Backup abort initiation*					
T + 78.0	Initiation of tumbling	3.17	377.0	83,000	53,000	43.0
T + 87.0	Canard deployment	2.386	106.0	97,208	71,912	34.9
T + 124.0	Apogee	1.45	15.0	120,000	131,000	0
T + 164.5	Termination of tumbling	1.60	53.0	95,000	189,000	-44.0
T + 245.2	LES plus BPC jettison	0.484	134.8	24,000	220,686	-87.4
T + 245.6	Forward heat shield ejection	0.47	130.0	23,800	220,800	-88.0
T + 247.2	Drogue chute deployment	0.42	87.0	23,500	221,000	-86.0
T + 248.0	Drogue line stretch*					
T + 256.0	Drogue disreef*					
	Pilot chute deployment					
T + 290.0	Line stretch (main chutes)*	0.21	43.0	11,000	221,000	-80.0
T + 298.0	Disreef (main chutes)*					
T + 582.0	Command module touchdown	0.2	9.0	4,036	221,000	-90.0

\*Information will be available at a later date.

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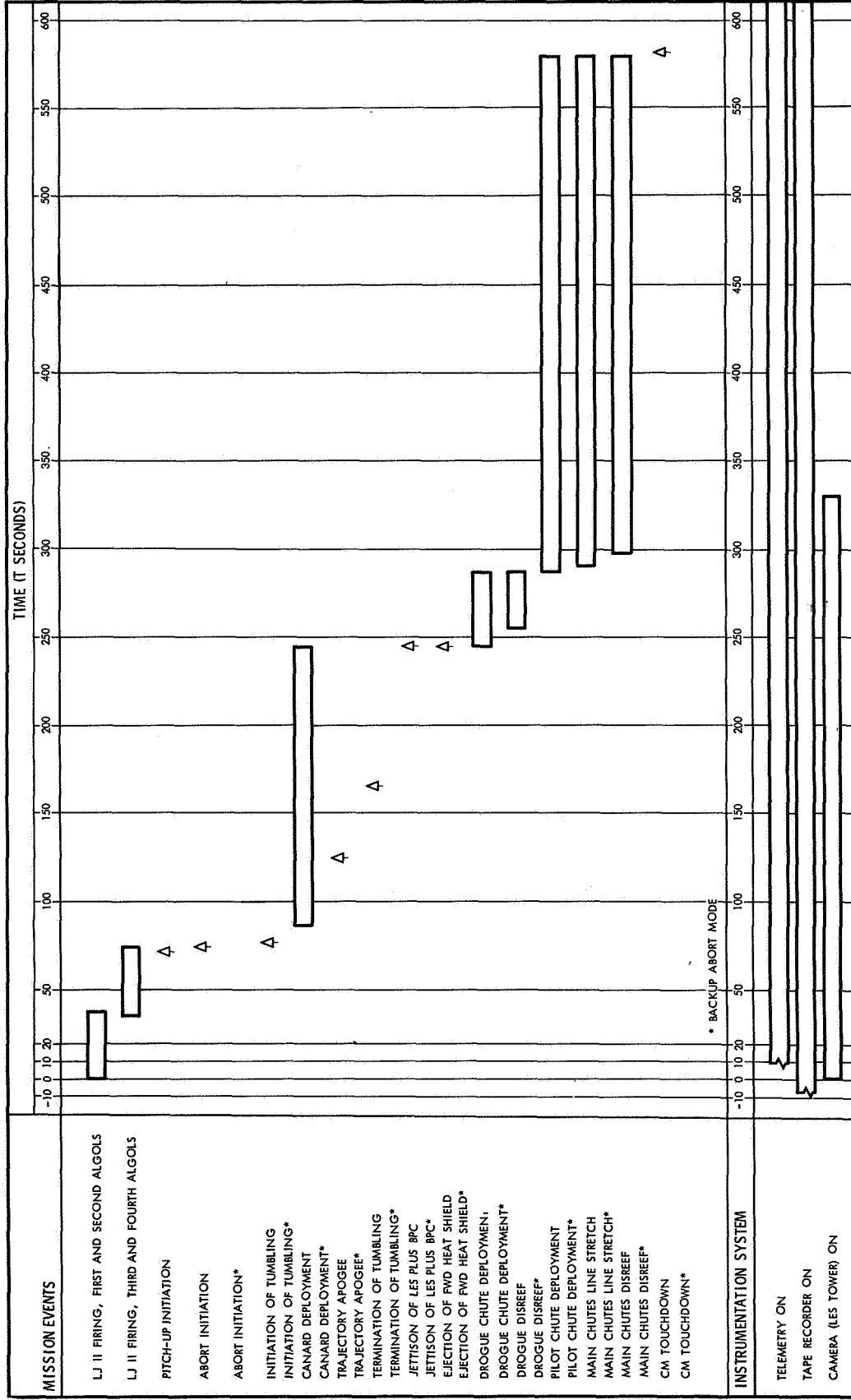


Figure 10-5. Operational Preliminary Time Lines

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SID 64-2174



## 11.0 POSTFLIGHT OPERATIONS

### 11.1 RECOVERY PERSONNEL AND EQUIPMENT REQUIREMENTS

Three personnel teams will be organized, equipped with appropriate support vehicle and tools, to examine and recover the command module at the touchdown site, the launch escape subsystem at the impact site, and the Little Joe II booster at its impact site. Each team will comprise a team leader, spotters, photographers, flight vehicle system specialists, technicians, firefighters, and transportation vehicle operators. The general positioning of the teams prior to launch will be coordinated by the WSMR recovery officer and the NASA recovery coordinator, based on continuously updated trajectory information from the control center during the countdown.

Equipment to be allocated to the recovery teams includes light aircraft or helicopter, ground transportation vehicles, firefighting vehicles, communication equipment, photographic equipment, propellant detectors, hardware detectors, protective clothing, pyrotechnic de-arm units, slings and dollies, containers, safety markers and flags, and measurement tools

### 11.2 EXAMINATION AND RECOVERY PROCEDURE

Upon arrival at the touchdown or impact site, photographers will approach cautiously and photograph the command module or flight system hardware from safe vantage points on the ground and in the air, showing attitudes and the relative position of the command module and parachutes or of the launch escape subsystem or Little Joe II components. Safe distances should be maintained until all unfired pyrotechnics have been flagged and de-armed and vehicle power has been switched off. The team will then begin a careful on-scene inspection of all hardware, and findings will be recorded in log books. Examination will be general in nature without requiring test equipment or extending recovery time appreciably. A reasonable attempt will be made to recover buried components of flight hardware. Depth of impressions will be measured and documented photographically. Damage incurred during recovery will be noted.





## 12.0 APOLLO DATA REQUIREMENTS

### 12.1 ONBOARD DATA

This section defines the onboard instrumentation and contains a summary list of measurements required for the evaluation of the test.

#### 12.1.1 Data Acquisition System

The methods of acquiring test vehicle instrumentation data are detailed here in a block diagram of the system (Figure 12-1).

##### 12.1.1.1 Onboard Tape Recorders

The two onboard recorders, which are primary data gathering devices, will be used to record telemeter outputs and measurements requiring high-frequency response. Each tape recorder unit will consist of a tape transport and recorder electronics. Tape recorder 2, in addition to the preceding, will utilize a tape recorder modulation package. The tape transports will operate at 15 inches per second and will be capable of providing approximately 10 minutes of recording time.

##### 12.1.1.2 Telemeter and Antenna Subsystem

The telemeter will be a PAM FM/FM subsystem consisting of two modulated transmitters, two 90-by-10 commutators, and an RF multiplexer. Each transmitter has a power output of approximately 5 watts. The telemeter antenna subsystem will consist of the VHF portion of the -Z VHF/2-kmc scimitar antenna located at  $X_c 23.253$  degrees (the +Z VHF/2-kmc scimitar antenna is an inactive antenna). There will be no in-flight calibration. Calibration will be performed prior to launch in three steps of 0, 2.5, and 5 volts consecutively. The required timing and accuracy, as well as format, will be compatible with type-B code system (IRIG 104-60, Time Format Standards).

##### 12.1.1.3 End Instruments and Signal Conditioning

End instruments will consist of the following: accelerometers, current monitors, position transducers, pressure transducers, rate transducers, strain gauges, temperature transducers, and acoustic transducers. Signal conditioning will consist of bridge adjustment units, thermocouple

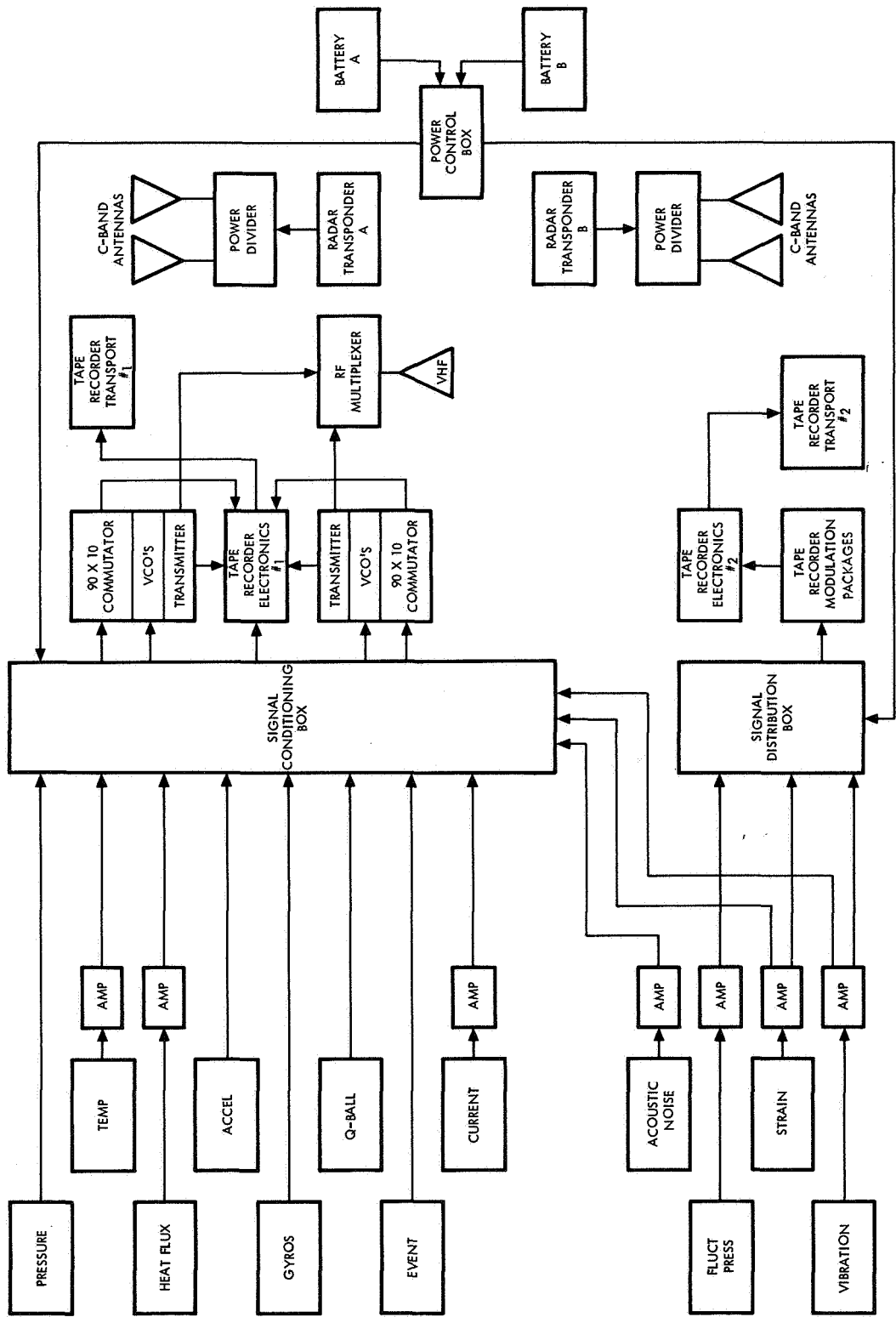


Figure 12-1. Onboard Instrumentation Network

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compensators, and phase-sensitive demodulators. These devices will condition the information received from the sensors into a modulation voltage for the subcarrier oscillators. The amplifier portion can be remotely calibrated for both R (range = 85 percent full scale) and Z (zero = 15 percent full scale).

#### 12.1.1.4 C-Band Transponder

Two C-band transponders receiving at 5480 mc and transmitting at 5700 mc will be installed to permit accurate radar tracking. Each transponder will operate independently and will be power-divided into two cavity antennas. The four cavity-backed helix antennas will be located 90 degrees apart at X<sub>c</sub>71. The C-band radar transponders respond to incoming pulse code signals and reply to tracking stations. Reply signals have a pulse rate frequency (PRF) that is a function of the interrogation rate of the tracking stations.

#### 12.1.1.5 Camera Installation

One camera will be installed in the launch escape vehicle. It will consist of a 16-millimeter, high-speed cine DBM-10 camera, 5-ampere-hour battery, 5.4-millimeter F2 lens, camera control box, and timing generator. The camera will be mounted in the launch escape tower along the centerline to permit photographic coverage of the boost protective cover as well as operation of the launch escape tower from the command module (see Figure 12-2). Information as to the frame rate, duration of camera operation, and start and stop time within the mission profile will be available at a later date.

#### 12.1.2 Instrumentation

A complete list of Spacecraft 002 measurements is contained in Appendix B. The Appendix B measurement requirement list consists of all flight measurement parameters by channel assignment. The official list is contained in Apollo Measurement Requirements, Spacecraft 002, SID 63-502, 1 December 1964.

Table 12-1 presents a summarization grouping of Spacecraft 002 onboard instrumentation by functional system, classification, and module location.

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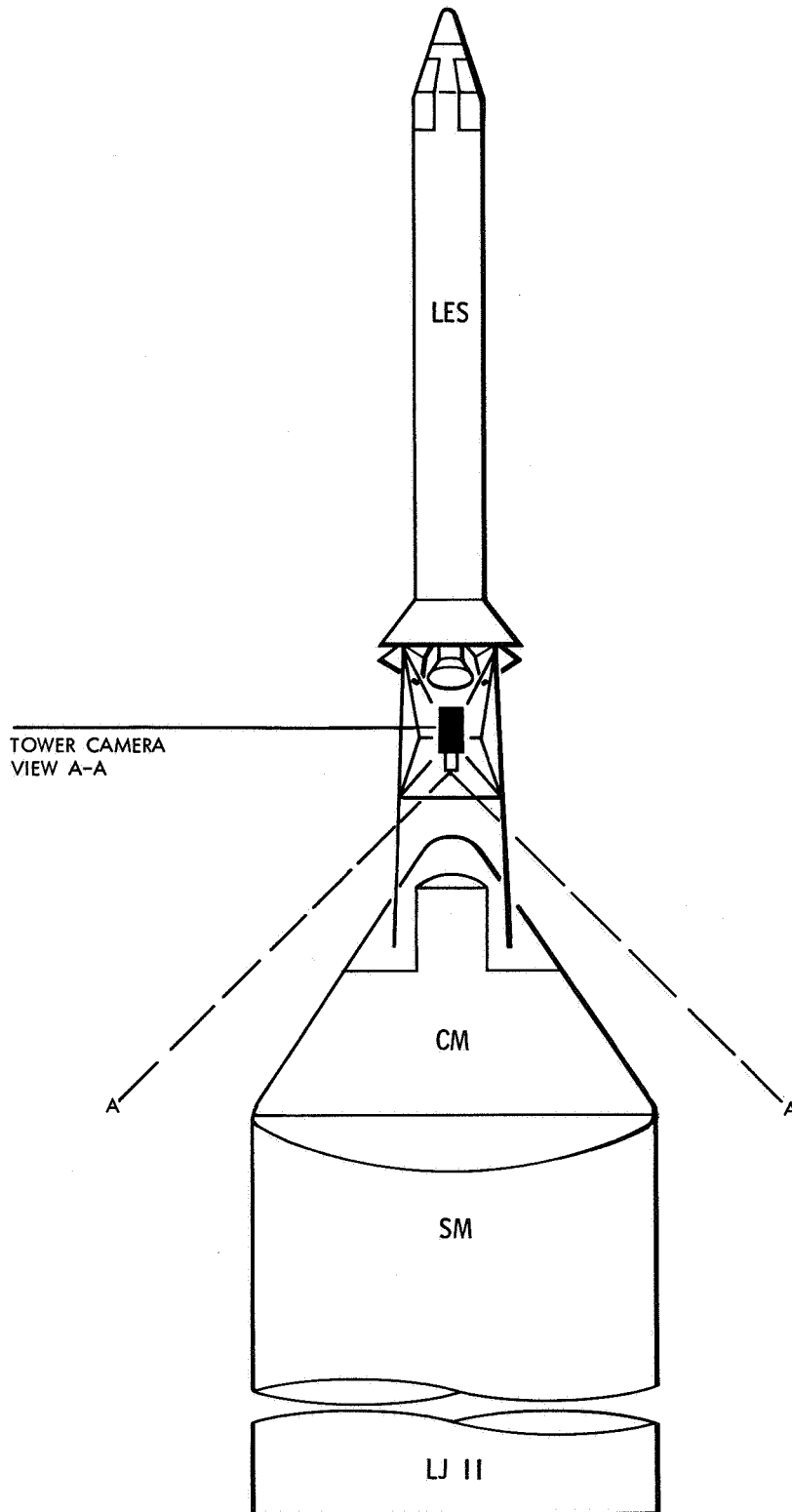
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Figure 12-2. Onboard Camera Location

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Table 12-1. Onboard Instrumentation Summary

Instrumentation	Range
<b>Structures</b>	
<u>Acceleration</u>	
Command module	
X axis high (1)	-10 to +20 g
X axis low (1)	-2 to +2 g
Y axis (1)	-5 to +5 g
Z axis (1)	-5 to +5 g
Launch escape tower	
Y axis (1)	-5 to +5 g
Z axis (1)	-5 to +5 g
<u>Vibration</u>	
Command module	
Display console, X, Y, Z (3)	-100 to +100 g
Lower equipment bay support, X, Y, Z (3)	-100 to +100 g
Normal to heat shield, floor, forward sidewall (3)	-100 to +100 g
Command module, X, Y, Z (3)	-100 to +100 g
Service module	
Sector II, III, IV, V, VI (3, 2, 1, 5, 2)	-500 to +500 g
RCS panel and support (2)	-500 to +500 g
RCS nozzle subsystem D (2)	-500 to +500 g
<u>Force</u>	
Launch escape tower	
Canard actuator link, $\pm Y$ (2)	$\pm 15,000$ lb
Tower leg load (4)	-15,000 to 50,000 lb



Table 12-1. Onboard Instrumentation Summary (Cont)

Instrumentation	Range
<u>Position</u>	
Launch escape tower	
Canard actuator displacement (1)	0 to +60 deg
<u>Pressure</u>	
Command module	
Conical surface (42)	+2 to +22 psia
Base pressure (4)	0 to +15 psia
Interior pressure (1)	0 to +15 psia
Service module	
Fluctuating, RCS subsystem D (1)	-3 to +3 psi
Fluctuating, (1)	-3 to +3 psi
Interior pressure (1)	0 to +15 psia
<u>Rate</u>	
Command module	
Heat flux (calorimeter) (6)	0 to +100 Btu per square foot per second
<u>Strain</u>	
Command module	
Axial hatch beam, in, out, vertical (2)	-5000 to +5000 microinches per inch
Tower longeron (6)	-5000 to +5000 microinches per inch



Table 12-1. Onboard Instrumentation Summary (Cont)

Instrumentation	Range
Forward longeron (12)	-5000 to +5000 micro-inches per inch
Side heat shield (8)	-5000 to +5000 micro-inches per inch
Service module	
Longitudinal and circumferential, bending, tension, and compression (20)	-7000 to +7000 micro-inches per inch
Tension-tie bolt (3)	0 to +5000 microinches per inch
Spacecraft compression pad (12)	-5000 to +5000 micro-inches per inch
<u>Temperature</u>	
Command module	
Calorimeter (6)	+32 to +482 F
Interior (1)	+32 to +302 F
Heat shield surface (6)	+32 to +302 F
Aft heat shield surface (2)	+32 to +302 F
Pressure hull (2)	+32 to +302 F
Service module	
Interior (1)	+32 to +302 F
Skin (4)	+32 to +482 F
Electrical power subsystem	
<u>Current</u>	
Command module	
Total instrumentation current (1)	0 to 50 amp
Battery A and B (2)	0 to 50 amp



Table 12-1. Onboard Instrumentation Summary (Cont)

Instrumentation	Range
<u>Voltage</u>	
Command module	
Instrumentation bus A and B (2)	+22 to +32 vdc
Logic bus A and B (2)	0 to +36 vdc
Launch escape subsystem	
<u>Pressure</u>	
Launch escape tower	
Pitch control and escape motor chamber pressure (2)	0 to 2000 psia
<u>Voltage</u>	
Command module	
LES pyro bus A and B (2)	0 to 36 vdc
Sequencer start signal A and B (2)	0 to 36 vdc
<u>Event</u>	
Command module	
Abort initiate relay close A and B (2)	Step
CM-SM separation relay close A and B (2)	Step
ELS sequencer A and B start relay close A and B (4)	Step





Table 12-1. Onboard Instrumentation Summary (Cont)

Instrumentation	Range
Tower jettison and separation relay close A and B (2)	Step
Canard deploy A and B relay close A and B (4)	Step
Forward heat shield jettison A and B (2)	Step
Backup abort timer close A and B (2)	Step
Abort initiate relay K-20, K-19, close A, B (2)	Step
Abort enable GSE system A and B (2)	Step
LES/pitch control motor, five relay close A and B (2)	Step
Earth Landing Subsystem	
<u>Pressure</u>	
Command module	
Barometric pressure static reference (1)	0 to 15 psia
<u>Event</u>	
Command module	
Drogue deploy relay close A and B (2)	Step
Main chute deploy, drogue release relay A and B (2)	Step
Baroswitch locking relay close A and B (2)	Step
Physical monitor drogue chute 1 and 2 (2)	Event



Table 12-1. Onboard Instrumentation Summary (Cont)

Instrumentation	Range
Flight technology	
<u>Position</u>	
Command module	
Attitude, pitch and yaw (2)	-180 to +180 deg
Attitude, roll, gyro out 1 and 2 (2)	-180 to +180 deg
Launch escape tower	
Angles of attack and sideslip, Q-ball (2)	-40 to +40 deg
<u>Camera</u>	
Launch escape tower	
Forward heat shield (1)	Looking aft, frame rate to be determined
<u>Pressure</u>	
Launch escape tower	
Dynamic pressure, Q-ball (1)	0 to +1250 psf
<u>Rate</u>	
Command module	
Rate, pitch, yaw, and roll (3)	-150 to +150 degrees per second
<u>Event</u>	
Command module	
Gyro segment switch 1 and 2 (2)	Step



Table 12-1. Onboard Instrumentation Summary (Cont)

Instrumentation	Range
<u>Acoustical</u>	
Command module	
Interior (1)	+110 to +150 decibels
Interior (1)	+100 to +140 decibels
Communications and instrumentation subsystem	
<u>Voltage</u>	
Command module	
Differentiated PDM (90 x 10 commutator) link A and B (2)	Voltage
Mixer 2 output link A and B (2)	Voltage
Sync A and B (90 x 10 commutator) link A and B (4)	+0 vdc
0 volt reference (90 x 10 commutator) link A and B (2)	+0 volts direct current
5-volt reference (90 x 10 commutator) link A and B (2)	+5 volts direct current
<u>Time</u>	
Command module	
Onboard timer	



## 12.2 EXTERNAL DATA

Tracking and support data requirements such as sampling rates, accuracies, and referencing of data are to be covered in SID 64-329, Apollo CSM Ground Operation Requirements Plan, Spacecraft 002. A summary of the general requirements are as follows.

### 12.2.1 Radar Tracking Data

Radar tracking data are required to provide azimuth, elevation, slant range, velocity, and acceleration. These data are required from launch to touchdown. Two C-band transponders will be provided in the command module to facilitate tracking. The booster and launch escape tower will be skin-tracked.

### 12.2.2 Photographic Coverage

#### 12.2.2.1 Sequential and Theodolite Tracking

Tracking will be required throughout the flight from liftoff at pad elevation (4036 feet) to apogee (120,000 feet) to command module touchdown 42 statute miles down range from the launch point. Cinetheodolite positioning must be accurate to plus or minus 20 feet and plus and minus one degree. See Table 12-2.

Table 12-2. Sequential and Theodolite Tracking

Flight Phase	Vehicle Unit	Cinematic Tracking	Cinetheodolite Positioning
Liftoff	Test vehicle stack	Coverage from 0 to 200 feet above the pad. Film speed and field of coverage shall be sufficiently great to document in detail contingent events.	Coverage from 0 to 700 feet above the pad to establish firmly the vehicle stack space position.
Launch (to abort point)	Test vehicle stack	Continuous tracking from approximately 100 feet above the pad up to the abort point (approximately 75,000 feet altitude). The field of	Continuous position recording immediately after liftoff to abort initiation.



Table 12-2. Sequential and Theodolite Tracking (Cont)

Flight Phase	Vehicle Unit	Cinematic Tracking	Cinetheodolite Positioning
Launch (to abort point) (Cont)	Test vehicle stack	coverage must permit detail observation of vehicle pitch maneuver at around 70,000 feet and behavior of the boost protective cover throughout the launch.	
LEV abort and descent	LJ II	Continuous coverage of the LJ II booster from just prior to CM-SM separation to 15 seconds after separation.	Continuous position recording from just prior to CM-SM separation to 15 seconds after separation.
	LEV	Continuous coverage of the LEV from just prior to CM-SM separation to LES jettison. Field of coverage and film speed must be appropriate for observation of canard deployment and boost cover behavior in detail. The canards deploy from full closed to full open in approximately 1/4 second.	Continuous position recording from just prior to CM-SM separation to LES jettison is required to establish rate of vehicle tumbling, stabilization, and orientation. Vehicle apogee (approximately 120,000 feet) and separation distance from the Little Joe II will also be established. Coverage is required.
LES jettison and CM deceleration	LES	Continuous coverage of the LES from just prior to jettison to 15 seconds after jettison will be required. Field of coverage and film speed must be appropriate for detailed observation of LES separation plus forward heat shield ejection (approximately 25,000 feet).	Continuous position recording from just prior to LES jettison to 15 seconds after jettison is required.



Table 12-2. Sequential and Theodolite Tracking (Cont)

Flight Phase	Vehicle Unit	Cinematic Tracking	Cinetheodolite Positioning
LES jettison and CM deceleration (Cont)	CM	Continuous coverage of the CM from just prior to LES jettison to 15 seconds after CM touchdown. Field of coverage and film speed must be appropriate for observation of parachute series deployment and behavior in detail with the CM.	Continuous position recording of the CM from just prior to LES jettison to CM touchdown is required to establish CM stabilization and rate of descent with the parachute subsystem.

The operational readiness requirements of sequential and theodolite tracking stations are presented in Appendix B.

#### 12.2.3 Documentary Films

A complete motion picture and still-photography documentary record of all key events associated with the Spacecraft 002 test will be required. Photographic coverage will include transportation, preflight preparations and checkout, flight operations, and postflight activities and events.

#### 12.2.4 Meteorological Data

The White Sands missile range (WSMR) will be required to furnish weather data consisting of the following parameters:

Parameter	Tolerance
Temperature	±2 F
Pressure	±1 percent
Wind velocity	±2 mph
Wind direction	±5 degrees
Density	±1 percent
Refractive index	±15 percent
Humidity	±1 percent



These data will be required from the White Sands, desert, and small missile range sites in 3-day general forecasts to support test scheduling and one hour before, during, and one hour after the test to support data reduction. The test support weather data will be required for various altitudes up to 120,000 feet. It should be noted that uninterrupted camera tracking is required to 120,000 feet mean sea-level, and up to 42 miles in range. Any weather phenomena that would inhibit these conditions will cause postponement of the test.

## 12.3 DATA ACQUISITION

### 12.3.1 Flight Telemetered Data

The flight telemetry data will be recorded by at least two White Sands missile range recording stations. The flight data will also be recorded by the telemetry trailer as a backup. The telemetry trailer data will be utilized as the source of quick-look data and on-site analysis. Each range recording station will produce three magnetic tapes, an original and two copies. Immediately after termination of the test, the original tape from each site will be given to NASA, one copy will be given to an S&ID Apollo Data Engineering representative, and the remaining copy will be retained by the White Sands missile range.

### 12.3.2 Onboard Magnetic Tape

As soon as possible after termination of the flight, the onboard tapes will be recovered and delivered to the telemetry trailer for copying. One master copy will be made from each of the original onboard tapes. Three additional copies will be made from each master copy. The original tapes and one copy will be delivered to NASA; S&ID will retain one copy at the field site; and the master and remaining copy will be given to an S&ID Apollo Data Engineering representative. All tape-recorded data (range and onboard copies) will be hand-carried to S&ID, Downey, by the Data Engineering representative within 24 hours after the flight.

### 12.3.3 Radar Data

Raw radar data, consisting of time, azimuth, elevation, and slant range of each of the three vehicle components (launch escape subsystem, command module, and service module) will be processed by White Sands missile range facilities into a Cartesian coordinate system using X, Y, and Z as space position axes. The origin of this system will be at the launch pad. The X axis will be positive in the direction of the intended flight path (north), the Y axis will be positive 90 degrees clockwise from the positive X axis (east), and the Z axis will be positive upward to form a left-handed orthogonal set. Magnetic tapes, compatible with an IBM 7090 computer,



containing time and X, Y, and Z position data for each vehicle component will be supplied to S&ID, Downey, within 48 hours after the flight. These data will be listed at Downey and examined for accuracy and validity. After the data quality is verified, the preliminary position data will be converted to a right-hand Cartesian coordinate system for further processing in the Downey computers, and all the required calculations will be performed. These data will then be processed into graphical form. The White Sands missile range will continue to process the position data into final format after it has supplied S&ID, Downey, with the preliminary position data.

#### 12.3.4 Optical Data

Both radar and optical (photographic) tracking of all vehicle units will be conducted throughout the flight. Radar position data will be the primary source of point mass trajectory information, and optical data will be the primary source of attitude information. Painted patterns on the launch escape vehicle will aid in optical data acquisition.

#### 12.3.5 Calibration Data

System calibrations for each instrumented parameter will be conducted in the Downey test preparation area. The results obtained from these calibrations will be computer processed to establish a best curve fit calibration curve. These calibration curves will be plotted, and a complete set of calibration curves will be maintained in Data Engineering and also transmitted with the vehicle to the field test site.

During vehicle checkout in the field, the calibrations will be checked against the Downey curves. Any measurement found to be out of limits will be rechecked, and the end instrument will be replaced as necessary to bring the measurement within required limits. Whenever an end instrument is changed, it will be replaced with a spare, and the calibration of the spare will be verified. Information regarding all end instrument changes will be immediately forwarded to Downey, where the calibration curves will be plotted, and the calibration will be recorded.





## 13.0 TEST REPORTS

### 13.1 MSC REPORTS

The analysis of the spacecraft mission and the writing of the postlaunch report will be completed at MSC, Houston, Texas by an analysis and report (A&R) team. This effort will be supported by MSC and contractor personnel.

#### 13.1.1 One-Hour Report

The MSC one-hour report will be an unclassified TWX prepared by the one-hour report group, approved by MSC-ASPO manager (PA) and MSC-FO manager (HA) and issued to NASA management approximately one hour after lift-off. The report will contain a brief notification of the time and place of the launch, and pertinent comments relating to the success or failure of the mission.

#### 13.1.2 Flight Status Report (48 hour)

The MSC flight status report will be a confidential TWX prepared by the flight status report group, approved by the PO, and issued to NASA management within 48 hours after mission termination. This report will include subsystem performance analysis detail and significant occurrences based on analysis of the quick-look data. If required, additional flight status reports will be written at 24-hour intervals to report additional significant facts.

#### 13.1.3 Postlaunch Report

The postlaunch report is the primary effort for the A&R team. It will be a confidential document issued by MSC-ASPO for distribution within the NASA organization and to ASPO contractors twenty-one calendar days after launch. It is intended to be used by personnel concerned with the planning for future missions. The postlaunch report will present results from an evaluation of the data available to the team during the analysis and writing period at MSC-PO and MSC-Houston.

Postlaunch report supplements will be prepared as required. These reports will present the results of additional study and analysis, and will be



prepared by MSC and contractor personnel. The reports will be issued by MSC-ASPO and distributed to the same persons that receive the postlaunch report.

### 13.2 NAA-ATO FIELD REPORTS

It is planned that preliminary information be made available for internal use of S&ID, Downey, subsequent to the Spacecraft 002 flight in the form of 1-hour, 24-hour, and 5-day field reports.

#### 13.2.1 One-Hour Report

The one-hour report is based on information available immediately after the test operation. In general, these reports will include a brief summary of the countdown, the launch time, a brief description of the mission, and pertinent observations and comments by NAA personnel at the launch site.

#### 13.2.2 24-Hour Flight Report

The 24-hour report will include a brief description of the countdown and recovery operations, a preliminary statement on the accomplishment of spacecraft objectives, a description of flight events and trajectory, and initial results of the quick-look data review.

#### 13.2.3 Five-Day Report

The five-day report will include a more detailed coverage of the subjects contained in the 24-hour flight report. In addition, it will contain results of postrecovery examination, a preflight history, a problem summary, and a list of the test data received and expected from the flight.

### 13.3 NAA ENGINEERING REPORTS

#### 13.3.1 Checkout Tests

Engineering will prepare a summary report of operational test and checkout procedures performed on the spacecraft or associated equipment. These reports will contain a general description of the test, a listing of the engineering problems encountered and the corrective actions taken, and recommendations for future vehicles and operations. Reports covering combined and integrated system checkout will be transmitted to RASPO-Downey.



### 13.3.2 Flight Tests

Engineering will perform an analysis of spacecraft data from the flight and will prepare a documented and integrated evaluation of the results which will be published as an NAA internal document. The engineering evaluation sections of this document will be delivered to NASA-MSD for review and subsequent publication as one or more postlaunch report supplements. The first supplement will be published 60 days after the flight. Additional supplements, if required, will be published at 30-day intervals up to a period of 120 days after the flight.



## 14.0 RANGE AND PAD SAFETY

### 14.1 RANGE SAFETY

Ordnance handling, vehicle—test facility integrated circuitry, and trajectory dispersion limitations will be specified in the NASA—MSC/WSMR Safety Mannual, NASA-SM-1, dated January 1964.

### 14.2 PAD SAFETY

#### 14.2.1 Personnel

The launch pad, blockhouse, and rocket motor assembly areas will be designated as limited-access areas, and only those personnel required to perform the assigned tasks will be near the dangerous areas.

Personnel admitted to the limited-access area will be required to wear safety helmets, safety shoes, and safety glasses as required. Hospital and medical care is to be provided by the WSMR medical staff. An ambulance and a fire truck with crews will be required to stand by during all operations with ordnance equipment.

#### 14.2.2 Explosive Control

The launch escape vehicle motors and other explosives will be stored by WSMR until needed in the hazardous assembly area or at the launch site. Ordnance simulators will be utilized in all checkout procedures where possible.

All explosives required in the Little Joe II will be stored by WSMR. A separate area will be designated for rocket motor buildup. The rocket motors will be transported by WSMR to the launch site as required for vehicle buildup. At all times, except for the checkout procedure, the igniter-squibs will have shorting plugs installed, and the rocket motor case will be grounded. During vehicle buildup, the service tower will be provided with lightning protection, and the vehicle will be connected directly to an earth ground to prevent static electricity buildup.

#### 14.2.3 Standard Operating Procedures

Standard operating procedures will be prepared by NASA as directed by Annex B, WSMR CIR NR385-2, Reference 5, and will conform to the



requirements of the "Ordnance Safety Manual," ORD M7-224, Reference 6. The standard operating procedures will define safety measures to be observed during the handling of the rocket motors, installation of igniters, testing of ignition circuits, installation of destruct charges and associated circuitry, utilization of shorting plugs, and final arming of the rocket motors and other ordnance devices.



## 15.0 GROUND SUPPORT EQUIPMENT

### 15.1 GSE REQUIREMENTS

Table 15-1 GSE specifies the units required for handling, checkout, and testing Spacecraft 002 in the test preparation area, the field hazardous assembly area, and the field launch site. Five groups of GSE will be used for handling, checkout, and testing:

1. Checkout equipment, C14-000, consisting of systems control and monitoring devices, interconnecting cables, and power distribution systems
2. Handling equipment, H14-000, consisting of slings, dollies, and workstands
3. Auxiliary equipment, A14-000, consisting of protective covers, warning streamers, simulators, etc.
4. Servicing equipment, S14-000, consisting of leak test units, fluid transfer units, etc.
5. Government furnished parts, GFP, consisting of special equipment

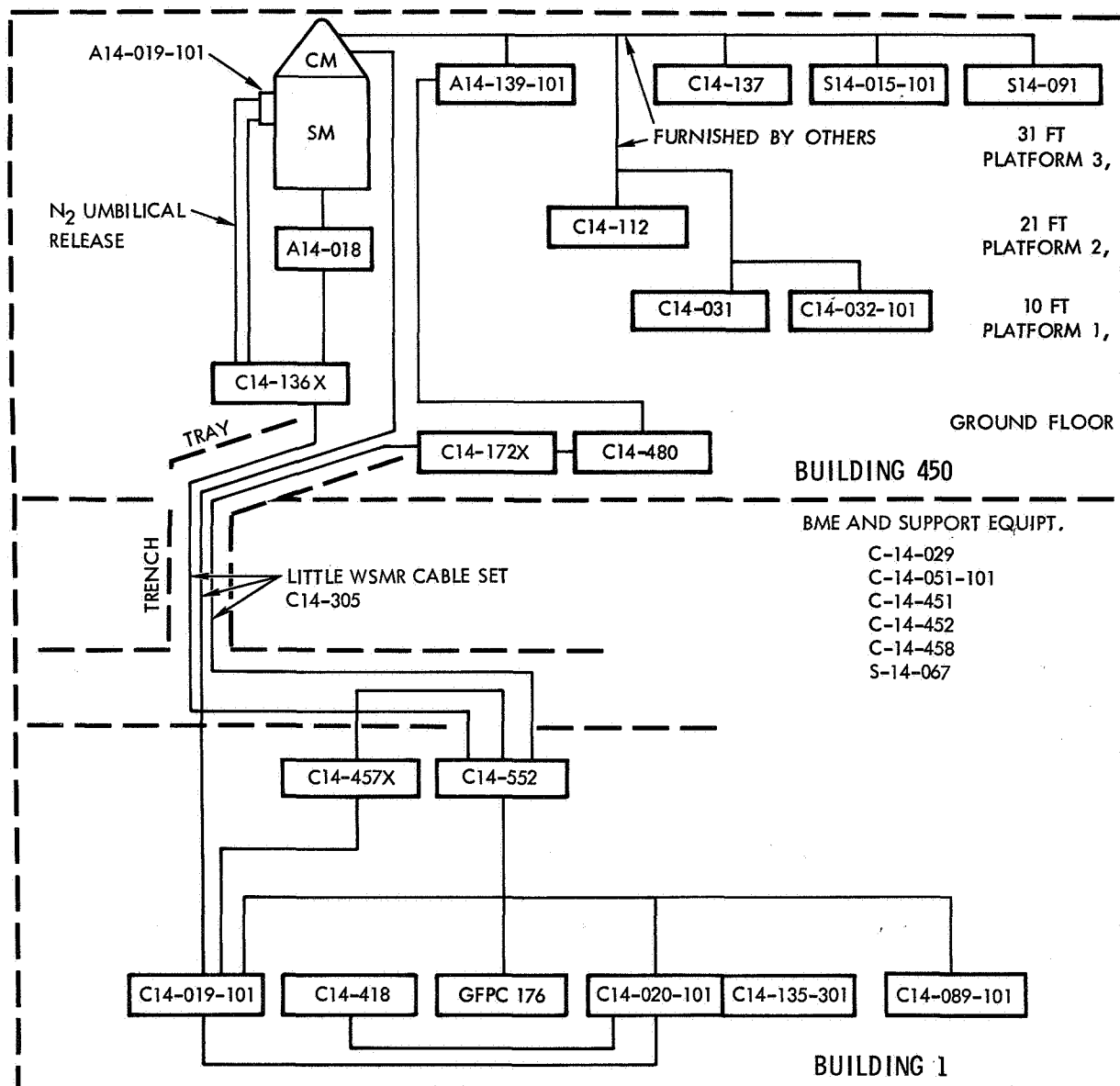
### 15.2 GSE LOCATIONS

Figure 15-1 shows location and configuration of GSE used for testing at Downey. Figure 15-2 shows the GSE location and configuration of the equipment to be used for checkout and launch at WSMR.

### 15.3 FIELD SITE SUPPORT REQUIREMENTS

The support requirements for the field site are as follows:

1. MA-8, or equivalent, ground cooling cart
2. Forklift, 5-ton capacity
3. Tractor, 2500-pound drawbar pull, gas-driven



## GSE EQUIPMENT INTERFACE

A14-018 Little Joe II Substitute Unit  
 A14-019-101 Disc Set-Umbilical, Fluid and Electrical  
 A14-139 PYRO Initiators Substitute Unit  
 C14-019-101 Test Condition Console  
 C14-020-101 Data Recording Group  
 C14-031 Onboard Recorder Checkout Unit  
 C14-032-101 Antenna Checkout Group  
 C14-089-101 High Response Recording Unit  
 C14-112 C-Band Transponder Checkout Unit  
 C14-135-301 Signal Conditioner  
 C14-137 Test Fixture Q-Ball (Analyzer)  
 C14-418 Spacecraft Ground Power Supply  
 C14-461 Breakout Box Set EMI Test (Interface not yet defined)  
 C14-480 Initiators Stimuli Unit

S14-015-101 Battery Charging Unit  
 S14-081 FDS (Building 450) N<sub>2</sub> Lines Only  
 S14-091 Battery Conditioner  
 GFPC-176 Apollo Instrumentation Checkout Console

## GSE J-BOX &amp; CABLE SET

C14-136X Umbilical J-Box  
 C14-172X Pad J-Box  
 C14-457X Black House J-Box  
 C14-305 Little WSMR Cable Set

Figure 15-1. GSE Requirements at Downey

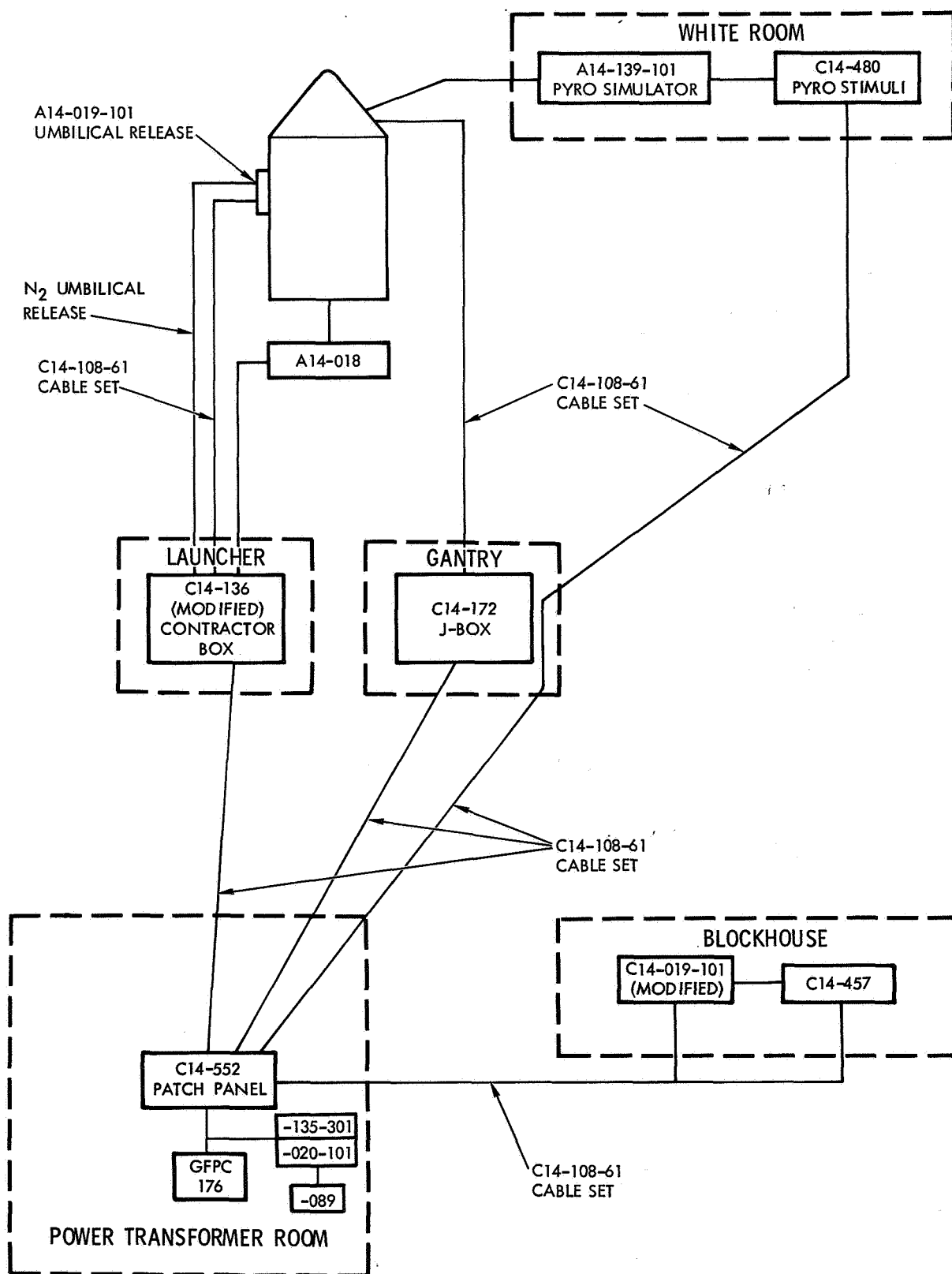


Figure 15-2. GSE Requirements at WSMR





4. Flatbed trailer, 10 by 40 feet (transport vehicle and equipment)
5. High-lift mobile aerial tower (60 feet per minute)
6. Heavy duty tug
7. Mobile crane, 10-ton capacity, 60-foot height
8. Flatbed truck, cross-country mobility, MMI type, required for post-flight recovery operations
9. Mobile crane, cross-country mobility, 10-ton capacity, 20-foot hook height, required for post-flight recovery operations
10. Maintenance and storage van, 35-foot (launch area)
11. Three mobile offices, 35-foot trailers (launch area)

Table 15-1. Required GSE Units

Model No.	Unit Description	Utilization	
		Downey	WSMR
A14-007	LES optical alignment set		X
A14-018	Launch vehicle substitute unit	X	X
A14-019-101	Disconnect set (electrical umbilical, fluid)	X	X
A14-026-801	Cap and plug set	X	X
A14-035-0002	Vacuum cleaner	X	
A14-036	Ground air circulating unit	X	
A14-036-301	Ground air circulating unit		X
A14-038	Launch escape motor—dummy	X	
A14-039	Launch escape jettison motor—dummy	X	
A14-040	Launch escape pitch control motor—dummy	X	
A14-046	Crane control, auxiliary	X	X
A14-047	Box level	X	X
A14-048	ELS equipment weight and balance set	X	
A14-128	Parachute installation kit	X	X
A14-130	LES tower explosive bolt wrench set	X	X
A14-132	Crew hatch command module handle	X	X
A14-134	Hydra set, 10-ton	X	X
A14-139-101	Pyro initiator substitute unit	X	X
A14-151	Forward heat shield installation kit	X	X



Table 15-1. Required GSE Units (Cont)

Model No.	Unit Description	Utilization	
		Downey	WSMR
A14-154	D-c digital indicator	X	X
A14-161	Launch escape motor inspection kit		X
A14-176	Service module cover	X	X
A14-180	Command module cover	X	X
A14-183	Tool set, canard activator	X	X
C14-019-101	Test conductor group	X	X
C14-020-101	Data recording group	X	X
C14-021	Telemetry ground station	X	
C14-029-101	LES sequencer BME	X	X
C14-031	Onboard recorder checkout unit	X	X
C14-032-101	Antenna checkout group	X	X
C14-051-101	Pyrotechnic BME	X	X
C14-089	Event recording unit, high response		X
C14-089-101	Event recording unit, high response	X	
C14-108-61	Electrical cable set, WSMR		X
C14-112	C-band radar transponder checkout unit	X	X
C14-135-301	Signal conditioner	X	X
C14-136	Umbilical junction box		X
C14-136X	Umbilical junction box	X	
C14-137	Test fixture Q-ball (analyzer)	X	X
C14-172	Junction box assembly		X
C14-172X	Junction box assembly	X	
C14-174	T/M kit		X
C14-305	Little WSMR cable set	X	
C14-418	Spacecraft ground power supply	X	
C14-451	Baroswitch test unit	X	X
C14-452	ELS sequencer BME	X	X
C14-457	Terminal board installations, blockhouse J-box		X
C14-457X	Terminal board installations, blockhouse J-box	X	
C14-458	Static EMI checkout device	X	X
C14-461	Breakout box set, EMI test		X
C14-480	Initiators stimuli unit	X	X
C14-552	Terminal distributors	X	X



Table 15-1. Required GSE Units (Cont)

Model No.	Unit Description	Utilization	
		Downey	WSMR
H14-008	Support base assembly	X	X
H14-011	Launch escape alignment stand	X	X
H14-016	LES weight and balance fixture		X
H14-017	Weight and balance fixture	X	X
H14-018	Escape tower support	X	X
H14-021	GSE handling cart	X	X
H14-029	Sling, flow skirt		X
H14-040	Electrical weight kit, 3000-pound capacity		X
H14-041	Electrical weight kit, 30,000-pound capacity		X
H14-043-101	Sling, jettison motor		X
H14-044	Parachute, handling sling	X	X
H14-052	Positioning trailer, narrow base	X	X
H14-054	Jettison motor support		X
H14-055	Launch escape motor support		X
H14-056	Sling, heat shield	X	X
H14-059	Sling set, weight and balance	X	X
H14-073	Sling, spacecraft minus LES	X	X
H14-074-101	Sling, spacecraft plus LES		X
H14-083-101	Cradle, LES transport		X
H14-084-101	Adapter, rollover		X
H14-085-201	Sling, LES horizontal handling		X
H14-090-301	Stand, recovery area access		X
H14-093	Boatswain chair		X
H14-093-101	Boatswain chair	X	
H14-094	Sling, pitch control motor nosecone		X
H14-096	Hook, LES ballast pickup	X	
H14-097	Stand, access, LES buildup	X	X
H14-099	Pitch control motor wrench		X
H14-100	Platform, 13-to-20 foot access (GFP)	X	X
H14-101	Platform, 3-to-10 foot access (GFP)	X	X
H14-113	Sling, service module	X	X
H14-114	Rack, service module	X	X
H14-127	Cable set, service module adapter weight and balance	X	X
H14-136-101	Hoist beam, service module/spacecraft adapter	X	X
H14-145	Sling, shipping container		X
H14-150	Sling, LES	X	X



Table 15-1. Required GSE Units (Cont)

Model No.	Unit Description	Utilization	
		Downey	WSMR
H14-156	LES sling, horizontal handling	X	X
H14-177	Spacecraft weight and balance set	X	X
H14-178	Load, receiving device	X	X
H14-182	Forward compartment heat shield sling	X	X
H14-9015	Jack set, weight and balance	X	X
H14-9030-101	Base, service module and adapter support	X	X
H14-9076-101	General purpose dolly	X	X
S14-06A	Heat shield pressure unit	X	
S14-015-101	Battery charging unit	X	X
S14-078	Pressure cost, LES rocket motor		X
S14-081	Fluid distribution unit (Building 450)	X	
S14-090	Mobile pressure test unit (0 to 2000 psig)		X
S14-091	Battery conditioner	X	X
GFP-A-028	Optical alignment unit	X	X
GFP-C-176	Instrumentation checkout console	X	X
GFP-H-203	Positioning trailer	X	X



## 16.0 GENERAL REQUIREMENTS

General requirements consisting of structural design criteria, environment, handling and transport, electromagnetic compatibility, thermal analysis data, and aerodynamic stability curves, are set forth in the "Master Spacecraft Specification" (SID 63-313). Specific requirements for Spacecraft 002 are contained in the "Contract End-Item Specification" (SID 63-699).

### 16.1 STRUCTURAL DESIGN CRITERIA

The spacecraft will consist of launch escape subsystem, command module, and service module. It will be designed and constructed to withstand normal ground and flight loads. The spacecraft for this mission will comprise the launch escape subsystem, the command module, and the service module. Secondary structure and mounting provisions will be provided for the subsystems required to accomplish the mission objectives. Detailed design criteria will be contained in "Structural Loads and Criteria" (SID 64-183, ARM-6).

### 16.2 ENVIRONMENTAL REQUIREMENTS

Specific environments for this flight are given in the "Performance and Interface Specification" (SID 63-949).

### 16.3 HANDLING AND TRANSPORTATION REQUIREMENTS

The handling and transportation requirements for Spacecraft 002 will be described in the "Preparation for Delivery and Transport of Spacecraft 002" process specification that will be prepared at a later date. The primary method of transportation will be by air. Air transport minimizes transport time and any possible compromise of hardware reliability because of environmental factors. Truck transport will be considered as an alternate to ensure maximum flexibility.

### 16.4 ELECTROMAGNETIC COMPATIBILITY REQUIREMENTS

The electromagnetic compatibility requirements are documented in the "Electromagnetic Interference Control Plan for the Apollo Space System"



(MC999-0002B). Although Spacecraft 002 will not be equipped with all the Apollo spacecraft subsystems, the subsystems contained will be designed to meet these requirements. As the vehicle passes through field operations, minimal checking will be accomplished to confirm compatibility of the subsystems, including GSE.



## 17.0 REFERENCES

SID 62-109	General Test Plan (Vol. 1 through 5)
SID 62-585	Apollo Hardware Utilization List
SID 62-1408	Apollo Measurement Requirements
SID 63-313	Master Apollo Spacecraft Specification
SID 63-271	Weight, Balance, and Inertia Summary, Spacecraft Configurations
SID 62-1450	Ground Support Equipment Plan
SID 62-931	Requirements for Work and Resources at White Sands Missile Range
SID 62-700-3	Apollo Ground Operations Requirements
SID 63-139	Apollo Transportation and Handling Procedures
SID 64-183 (ARM-6)	Structural Loads and Criteria
SID 62-417	GSE Planning and Requirements
PRD	Program Requirements Document
Number to be assigned at a later date	Preparation for Delivery and Transport of Apollo Spacecraft
62-TO-695-322-1	GSE Engineering Planning Data
62-DI-695-420-1	Definitions and Objectives of Apollo Hard- ware Test Articles
ATO 63-43	Detail Test Plan, Boilerplate 6
WP1049B	Project Apollo Flight Mission Directive, Mission PA-1
ATO 63-125	Detail Test Plan, Boilerplate 12
MC999-0051A	Environmental Design and Test Requirements
MC999-0002B	Electromagnetic Interference Control Plan, the Apollo Space System
WP-1065A	Project Apollo Flight Mission Directive, Mission A-001
SID 63-949	Performance and Interface Specification, Apollo Spacecraft 002 and Little Joe II Launch Vehicle
V14-900102	SID Integrated System Schematic Drawing, Spacecraft 002, at Downey
SID 63-502	Apollo Measurement Requirements, Spacecraft 002
Number to be assigned at a later date	Postflight Recovery and Inspection, Spacecraft 002



MA-0201-0071	Integrated System Checkout Requirements, Spacecraft 002
SID 63-699	Vehicle Contract End-Item Detailed Specification, Spacecraft 002, Downey
Number to be assigned at a later date	Mission Data Plan, Spacecraft 002
NASA Program Apollo Working Paper 1135	Reporting Plan, Apollo Mission A-102 (Boilerplate 15)
SID 64-329	Apollo CSM Ground Operation Requirements Plan, Spacecraft 002





## APPENDIX A

## DEFINITIONS

## OBJECTIVE CATEGORY

The objectives for Apollo Mission A-004 (Spacecraft 002) are classified according to their importance: first order, second order, and third order. Definitions for these classifications were obtained from NASA formal letter PL3/2/M-64-442 to NAA, dated 21 July 1964. The NAA identification number for the letter is 9073MA.

First-Order

First-order test objectives are the main purpose for conducting the mission. The mission profile will be tailored to optimize the accomplishment of first-order test objectives.

Second-Order

Second-order test objectives consist of (1) those specific tests which constrain (must be completed satisfactorily prior to) succeeding flights and (2) flight proof tests of spacecraft equipment. The minimum spacecraft subsystem and instrumentation configuration required for the mission is established by the first- and second-order test objectives.

Third-Order

Third-order test objectives support or enhance succeeding Apollo flights, supply supplementary data for overall spacecraft evaluation, and accomplish scientific experiments assigned to the mission.

## OBJECTIVE TERMINOLOGY

The action terminology selected in the makeup of the individual objective denotes the extent of analysis required for accomplishment of that objective.



### Demonstrate

The term "demonstrate" denotes the occurrence of an action or an event during a test. The accomplishment of an objective of this type requires a qualitative answer. The answer will be derived through the relation of this action or event to some other known information or occurrence. This category of objective implies a minimum of airborne instrumentation, and/or that the information to be obtained external to the spacecraft.

### Verify

The term "verify" means to quantitatively demonstrate the safe functioning, achievement of minimum performance, and operational suitability of equipment. "Verify," also implies a quantitative investigation of overall operation which indicates quantitatively that the system does in fact operate as designed, but does not imply that the instrumentation should necessarily be adequate to allow performance deficiencies to be isolated. "Verify," therefore, and insofar as the instrumentation is concerned, would seem to be somewhat more than "demonstrate" but somewhat less than "determine"

### Determine

The term "determine" denotes the measurement of performance of any system or subsystem. This category implies a quantitative investigation of overall operation which includes, generally, instrumentation for measuring basic inputs and outputs of the system or subsystem. The information obtained should indicate to what extent the system is operating as designed. The instrumentation should allow performance deficiencies to be isolated to either the system or to the system inputs.

### Evaluate

The term "evaluate" denotes the measurement of performance of any system or subsystem as well as the performance and/or interaction of its components under investigation. The accomplishment of this type of objective requires quantitative data on the performance of both the system or subsystem and its components. The performance levels will then be analyzed for their contribution toward performance of the system. This category will provide the most detailed information of any of these categories.

### Obtain Data

The term "obtain data" denotes the gathering of engineering information to be measured to augment the general knowledge required in the development



of the overall spacecraft. This category also may be used for supplemental investigations, such as environmental and ground equipment studies. The degree of instrumentation is not implied by this definition.

## SYSTEMS PRIORITIES

The spacecraft systems are assigned either primary or secondary priorities. During the initial test program, when incomplete systems are tested, this definition will apply to the subsystem and component level.

### Primary

Primary subsystems are functionally required for the spacecraft to successfully accomplish first- and second-order test objectives. A positive indication of satisfactory operation of primary subsystems must be available before launch.

### Secondary

Secondary subsystems are not functionally required for the spacecraft to complete its planned mission (first- and second-order objectives). Mal-function of secondary subsystems will not require a mandatory countdown hold or scrub.

## SYSTEM OR SUBSYSTEM

### Complete (C)

The term "complete" designates a system or subsystem intended to fulfill manned mission requirements.

### Partial (P)

The term "partial" designates anything less than a complete system or subsystem of the type intended to fulfill manned mission requirements.

### Simulated (S)

The term "simulated" designates a nonfunctional system or subsystem that simulates the configuration.

### Interim (I)

The term "interim" designates a functional substitute system or subsystem.



## APPENDIX B

## MISSION RULES

The mission rules established herein for Apollo Power-On Tumbling Boundary Abort Mission A-004 with Spacecraft 002 govern the readiness of the test vehicle, the applicable systems of the Little Joe II booster, the WSMR test range, and the weather for satisfactory launch and mission accomplishment. The mission rules are divided into four categories: 1) Spacecraft 002 and Little Joe II systems required status, 2) Operational readiness required of individual onboard instrumentation units, plus the applicability of these units to the mission objectives, 3) test range support systems requirements, and 4) flight dynamics and weather requirements.

The terms and symbols that appear in the following tables are defined in the following list.

Term	Definition
Mandatory (M)	Hold to fix or scrub.
Highly Desirable (HD)	Hold to fix at discretion of operations director or test conductor.
Desirable (D)	Proceed without hold.

## SC 002 AND LJ-II SYSTEM OR COMPONENT

Table B-1. SC 002 and LJ II Operational Readiness

System or Component	Quantity	Operational Readiness	Limit
Launch Escape System			
Rocket motors		M	
Canards		M	
Q-ball		M	



Table B-1. SC 002 and LJ-II Operational Readiness (Cont)

System or Component	Quantity	Operational Readiness	Limit
Earth Landing System			
Mortars		M	
Parachutes		M	
Retention and Separation System			
CM-SM		M	
LES-CM		M	
Forward Heat Shield-CM		M	
Sequencer System			
Tower		M	
Mission		M	
Landing		M	
Electrical Power System			
Instrumentation batteries	2	M	28 volts minimum under load
Pyrotechnic batteries	2	M	32 volts minimum armed bus
Logic batteries	2	M	30 volts minimum with armed bus and 32 volts no load
Communication and Instrumentation System			
On-board tape recorders	2	M	
On-board camera		M	
Telemetry	2	M	
C-band Transpondors	2	M (1) HD (1)	



Table B-1. SC 002 and LJ-II Operational Readiness (Cont)

System or Component	Quantity	Operational Readiness	Limit
Launch Vehicle			
Abort initiation delay timer		M	
Abort backup timer		M	

## TEST VEHICLE INSTRUMENTATION

This section of the Mission Rules concerns only the operational readiness of the instrumentation units themselves and not that of the systems or equipment they monitor. For additional information involving instrumentation, refer to SID 63-502, Apollo Measurement Requirements for Spacecraft 002, dated 31 July 1964.

Table B-2 shows operational readiness of on-board instrumentation.



Table B-2. Operational Readiness of On-Board Instrumentation

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order							3rd Order					
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
TM A CONTINUOUS CHANNEL																		
CK0001R	Unassigned	A-1																
	Unassigned	A-2																
	Pitch rate gyro output	A-3	M															
	Unassigned	A-4																
	Unassigned	A-5																
CA0001A	X-axis spacecraft acceleration high	A-6	M															
CA0005A	Y-axis spacecraft acceleration	A-7	M															
CT0001W	On-board timer	A-8	M															
LA0012A	Z-axis tower acceleration	A-9	M															
LD0012P	Pitch control motor chamber pressure	A-10	HD															
SA0957P	Unassigned	A-11																
	Unassigned	A-12																
	Unassigned	A-13																
	Unassigned	A-14																
	SM fluctuating pressure	A-15	D															
CA2246D	RCS panel																	
	X-axis CM vibration	A-16	HD															
	Commutator 90 x 10	A-E	M															



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order					3rd Order							
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
CT0025V	0 V reference (90 x 10 commutator) link A	A-E-1	M															
CT0026V	5 V reference (90 x 10 commutator) link A	A-E-2	M															
CC0001V	D-C voltage main bus A (instrumentation)	A-E-3	M															
CC0002V	D-C voltage main bus B (instrumentation)	A-E-4	M															
	Unassigned	A-E-5																
	Unassigned	A-E-6																
	Unassigned	A-E-7																
	Unassigned	A-E-8																
CC0005C	Total direct-current (instrumentation)	A-E-9	HD															
CA0280P	Base pressure 1	A-E-10	HD									X						
CA0281P	Base pressure 2	A-E-11	HD									X						
CA0282P	Base pressure 3	A-E-12	HD									X						
CA0283P	Base pressure 4	A-E-13	HD									X						
	Unassigned	A-E-14																
	Unassigned	A-E-15																
CD0230X	Fwd heat shield jettison A	A-E-16	HD									X						





Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order					3rd Order							
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
CD00231X	Fwd heat shield jettison B	A-E-16	HD															
	Unassigned	A-E-17																
CD00005V	D-C voltage LES pyro bus A	A-E-18	M									X						
CD00006V	D-C voltage LES pyro bus B	A-E-19	M															
CD00023X	CM-SM separate relay	A-E-20	HD															
	close A																	
CD00024X	CM-SM separate relay	A-E-20	HD										X					
	close B																	
CD0190X	LES/pitch control motor	A-E-21	HD	X														
	fire relay close A																	
CD0191X	LES/pitch control motor	A-E-21	HD	X														
	fire relay close B																	
CD00033X	TWR jettison and separate	A-E-22	HD															
	relay close A																	
CD00034X	TWR jettison and separate	A-E-22	HD															
	relay close B																	
CD00008V	Sequencer start signal A	A-E-23	HD															
CD00009V	Sequencer start signal B	A-E-24	HD	X														
CD00025X	ELS sequencer A start relay	A-E-25	HD	X														
	close A																	
CD00026X	ELS sequencer A start relay	A-E-25	HD															
	close B																	
CD0116X	Canard deploy A relay	A-E-26	HD															
	close A																	



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order					3rd Order							
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
CD0117X	Canard deploy A relay close B	A-E-26	HD			X												
CD0118X	Canard deploy B relay close A	A-E-27	HD			X												
CD0119X	Canard deploy B relay close B	A-E-27	HD			X												
CD0021X	Abort initiate relay close A	A-E-28	HD	X														
CD0152X	Abort initiate relay K-20 close A	A-E-28	HD	X														
CD0022X	Abort initiate relay close B	A-E-29	HD	X														
CD0153X	Abort initiate relay K-19 close B	A-E-29	HD	X														
CD0027X	ELS sequencer B start relay close A	A-E-30	HD												X			
CD0028X	ELS sequencer B start relay close B	A-E-30	HD												X			
CD0150X	Backup abort timer close A	A-E-31	D	X														
CD0151X	Backup abort timer close B	A-E-31	D	X														
CD0154X	Abort enable GSE system A	A-E-32	HD															
CD0155X	Abort enable GSE system B	A-E-32	HD															
CE0001X	Drogue deploy relay close A	A-E-33	HD												X			
CE0002X	Drogue deploy relay close B	A-E-33	HD												X			
CE0003X	Main chute deploy-drogue relay A	A-E-34	HD												X			



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order					3rd Order						
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
CE0004X	Main chute deploy-drogue relay B	A-E-34													X			
CE0009X	Baro switch lockin relay close A	A-E-35	HD									X			X			
CE0010X	Baro switch lockin relay close B	A-E-35	HD									X			X			
CC0003V	D-C voltage logic bus A	A-E-36	M															
CC0004V	D-C voltage logic bus B	A-E-37	M															
LA0215G	Load tower leg	A-E-38	HD															
LA0216G	Load tower leg	A-E-39	HD															
LA0217G	Load tower leg	A-E-40	HD															
LA0218G	Load tower leg	A-E-41	HD															
CA1880S	Strain axis side heat shield horizontal out	A-E-42	HD						X							X		
CA1881S	Strain axis side heat shield horizontal in	A-E-43	HD						X							X		
CA1882S	Strain axis side heat shield vertical out	A-E-44	HD						X							X		
CA1883S	Strain axis side heat shield vertical in	A-E-45	HD						X							X		
CA1884S	Strain axis side heat shield horizontal out	A-E-46	HD						X							X		
CA1885S	Strain axis side heat shield horizontal in	A-E-47	HD						X							X		



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order					3rd Order						
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
CA1886S	Strain axis side heat shield vertical out	A-E-48	HD						X							X		
CA1887S	Strain axis side heat shield vertical in	A-E-49	HD						X							X		
CA1604S	Strain axis forward longeron 4 out vertical	A-E-50	HD						X	X						X		
CA1608S	Strain axis forward longeron 8 out vertical	A-E-51	HD						X	X						X		
CA1601S	Strain axis forward longeron 2 in vertical	A-E-52	HD						X	X						X		
CA1605S	Strain axis forward longeron 4 in vertical	A-E-53	HD						X	X						X		
CA1609S	Strain axis forward longeron 8 in vertical	A-E-54	HD						X	X						X		
CA1606S	Strain axis forward longeron 4 out vertical	A-E-55	HD						X	X						X		
CA1603S	Strain axis forward longeron 2 in vertical	A-E-56	HD						X	X						X		
CA1607S	Strain axis forward longeron 4 in vertical	A-E-57	HD						X	X						X		
CA0510S	Strain axis hatch beam out vertical	A-E-58	HD						X	X						X		
CA0511S	Strain axis hatch beam in vertical	A-E-59	HD						X	X						X		



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order							3rd Order				
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
SA2020S	Strain axis tension bolt beam 2	A-E-60	HD								X							
SA2021S	Strain axis tension bolt beam 4	A-E-61	HD								X							
SA2022S	Strain axis tension bolt beam 6	A-E-62	HD								X							
CA1600S	Strain axis forward longeron 2 out	A-E-63	HD						X	X						X		
CA1602S	Strain axis forward longeron 2 out	A-E-64	HD						X	X						X		
CA2617S	Strain axis tower longeron 8 gus side	A-E-65	HD						X	X						X		
CA2618S	Strain axis tower longeron 8 gus side	A-E-66	HD						X	X						X		
SA0876T	SM skin temperature outer	A-E-67	HD															
SA0877T	SM skin temperature inner	A-E-68	HD															
SA0878T	SM skin temperature outer	A-E-69	HD															
SA0879T	SM skin temperature outer	A-E-70	HD															
LK0023H	Angle of attack	A-E-71	M	X										X				
LK0025P	Dynamic pressure	A-E-72	HD															
LK0024H	Angle of sideslip	A-E-73	M	X														
CK0016H	Pitch attitude gyro output	A-E-74	HD	X										X		X		
CK0018H	Yaw attitude gyro output	A-E-75	HD	X										X		X		
CK0050H	Roll attitude gyro output 1	A-E-76	HD	X										X		X		



Table B-2. Operational Readiness of On-Board Instrumentation

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order					3rd Order						
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
CK0051H	Roll attitude gyro output 2	A-E-77	HD	X			X								X			
CK0052X	Gyro segment switch 1	A-E-78	D															
CK0053X	Gyro segment switch 2	A-E-79	D															
CA0610T	CM interior temperature	A-E-80	HD															
SA0612T	CM interior temperature	A-E-81	HD															
CA0611P	SM interior pressure	A-E-82	HD															
SA0613P	SM interior pressure	A-E-83	HD								X							
CA1077T	Temperature pressure hull	A-E-84	D															
CA1078T	Temperature pressure hull	A-E-85	D										X					
CE0035P	Barometric pressure static reference	A-E-86	M															
SA4001S	Strain spacecraft SM component pad beam 1A	A-E-87	HD								X							
SA4002S	Strain spacecraft SM component pad beam 1B	A-E-88	HD								X							
CT0027V	Sync A (90 x 10 commutator) link A	A-E-89	M															
CT0028V	Sync B (90 x 10 commutator) link A	A-E-90	M															



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order							3rd Order					
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
TM B CONTINUOUS CHANNEL																		
CK0002R CK0003R	Unassigned	B-1																
	Unassigned	B-2																
	Yaw rate gyro output	B-3	M	X												X		
	Roll rate gyro output	B-4	M	X												X		
CA0002A	Unassigned	B-5																
	Unassigned	B-6																
	X-axis spacecraft acceleration low	B-7	HD	X										X				
	On-board timer	B-8	M	X														
LA0011A CA0007A	Y-axis tower acceleration	B-9	M	X														
	Z-axis spacecraft acceleration	B-10	M	X										X			X	
LD0013P	Escape motor chamber pressure	B-11	M	X														
LA0200H	Canard actuator displacement	B-12	D															
	Unassigned	B-13																
LA0202G	Loads - Y canard actuator link	B-14	HD															
LA0201G	Loads +Y canard actuator link	B-15	HD															
CA2247D	Y-axis CM vibration	B-16	HD															
	Commutator B 90 x 10	B-E	M														X	



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order					2nd Order					3rd Order				
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
CT0500V	0 V reference (90 x 10 commutator) link B	B-E-1	M															
CT0501V	5 V reference (90 x 10 commutator) link B	B-E-2	M															
SA4003S	Strain spacecraft component pad beam 2A	B-E-3	HD							X								
SA4004S	Strain spacecraft component pad beam 2B	B-E-4	HD							X								
SA4005S	Strain spacecraft component pad beam 3A	B-E-5	HD							X								
SA4006S	Strain spacecraft component pad beam 3B	B-E-6	HD							X								
SA4007S	Strain spacecraft component pad beam 4A	B-E-7	HD							X								
SA4008S	Strain spacecraft component pad beam 4B	B-E-8	HD							X								
SA4009S	Strain spacecraft component pad beam 5A	B-E-9	HD							X								
SA4010S	Strain spacecraft SM component pad beam 5B	B-E-10	HD							X								
SA4011S	Strain spacecraft SM component pad beam 6A	B-E-11	HD							X								
SA4012S	Strain spacecraft SM component pad beam 6B	B-E-12	HD							X								





Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability															
				1 st Order		2nd Order							3rd Order						
				1	2	1	2	3	4	5	6	7	1	2	3	4	5		
CA1512S	Strain axis tower longeron 2 gus side	B-E-13	HD					X	X										
CA1513S	Strain axis tower longeron 2 gus edge	B-E-14	HD					X	X										
CA1514S	Strain axis tower longeron 4 gus side	B-E-15	HD					X	X										
CA1515S	Strain axis tower longeron 4 gus edge	B-E-16	HD					X	X										
CA1610S	Strain axis forward longeron 8 out vertical	B-E-17	HD					X	X										
CA1611S	Strain axis forward longeron 8 in vertical	B-E-18	HD					X	X										
CE0014X	Physical mon drogue chute 1	B-E-19	HD												X				
CE0015X	Physical mon drogue chute 2	B-E-20	HD												X				
	Unassigned	B-E-21																	
	Unassigned	B-E-22																	
	Unassigned	B-E-23																	
	Unassigned	B-E-24																	
	Unassigned	B-E-25																	
	Unassigned	B-E-26																	
CA0574R	Heat flux (calorimeter) hatch window	B-E-27	HD															X	
CA0645T	Calorimeter body temperature hatch window	B-E-28	HD															X	
CA0575R	Heat flux (calorimeter) 23	B-E-29	HD																X



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order					3rd Order						
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
CA0646T	Calorimeter body temperature 23	B-E-30	HD														X	
CA0576R	Heat flux (calorimeter) window	B-E-31	HD														X	
CA0647T	Calorimeter body temperature window	B-E-32	HD														X	
CA0577R	Heat flux (calorimeter) 22	B-E-33	HD														X	
CA0648T	Calorimeter body temperature 22	B-E-34	HD														X	
CA0578R	Heat flux (calorimeter) 21	B-E-35	HD														X	
CA0649T	Calorimeter body temperature 21	B-E-36	HD														X	
CA0579R	Heat flux (calorimeter) 20	B-E-37	HD														X	
CA0650T	Calorimeter body temperature 20	B-E-38	HD														X	
CA1069T	Temperature CM heat shield surface	B-E-39	HD														X	
CA1070T	Temperature CM heat shield surface	B-E-40	HD														X	
CA1071T	Temperature CM heat shield surface	B-E-41	HD														X	
CA1072T	Temperature CM heat shield surface	B-E-42	HD														X	
CA1073T	Temperature CM heat shield surface	B-E-43	HD														X	



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order					3rd Order							
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
CA1074T	Temperature CM heat shield surface	B-E-44	HD															X
CA1075T	Temperature aft heat shield surface	B-E-45	D															
CA1076T	Temperature aft heat shield surface	B-E-46	D															
CA0228P	Conical surface pressure 1	B-E-47	N1									X						
CA0229P	Conical surface pressure 2	B-E-48	N1									X						
CA0230P	Conical surface pressure 3	B-E-49	N1									X						
CA0232P	Conical surface pressure 5	B-E-50	N1									X						
CA0233P	Conical surface pressure 6	B-E-51	N1									X						
CA0234P	Conical surface pressure 7	B-E-52	N1									X						
CA0235P	Conical surface pressure 8	B-E-53	N1									X						
CA0237P	Conical surface pressure 10	B-E-54	N1									X						
CA0238P	Conical surface pressure 11	B-E-55	N1									X						
CA0239P	Conical surface pressure 12	B-E-56	N1									X						
CA0241P	Conical surface pressure 14	B-E-57	N1									X						
CA0242P	Conical surface pressure 15	B-E-58	N1									X						
CA0247P	Conical surface pressure 20	B-E-59	N1									X						
CA0248P	Conical surface pressure 21	B-E-60	N1									X						
CA0249P	Conical surface pressure 22	B-E-61	N1									X						
CA0250P	Conical surface pressure 23	B-E-62	N1									X						
CA0252P	Conical surface pressure 25	B-E-63	N1									X						
CA0254P	Conical surface pressure 27	B-E-64	N1									X						



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order					3rd Order							
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
CA0255P	Conical surface pressure 28	B-E-65	N1									X						
CA0257P	Conical surface pressure 30	B-E-66	N1									X						
CA0258P	Conical surface pressure 31	B-E-67	N1									X						
CA0259P	Conical surface pressure 32	B-E-68	N1									X						
CA0260P	Conical surface pressure 33	B-E-69	N1									X						
CA0262P	Spacecraft conical surface pressure 35	B-E-70	N1									X						
CA0265P	Spacecraft conical surface pressure 38	B-E-71	N1									X						
CA0266P	Spacecraft conical surface pressure 39	B-E-72	N1									X						
CA0267P	Spacecraft conical surface pressure 40	B-E-73	N1									X						
CA0268P	Spacecraft conical surface pressure 41	B-E-74	N1									X						
CA0269P	Spacecraft conical surface pressure 42	B-E-75	N1									X						
CA0270P	Spacecraft conical surface pressure 43	B-E-76	N1									X						
CA0271P	Spacecraft conical surface pressure 44	B-E-77	N1									X						
CA0272P	Spacecraft conical surface pressure 45	B-E-78	N1									X						
CA0273P	Spacecraft conical surface pressure 46	B-E-79	N1									X						



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order					3rd Order						
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
CA0274P	Spacecraft conical surface pressure 47	B-E-80	N1										X					
CA0275P	Spacecraft conical surface pressure 48	B-E-81	N1										X					
CA0276P	Spacecraft conical surface pressure 49	B-E-82	N1										X					
CA0277P	Spacecraft conical surface pressure 50	B-E-83	N1										X					
CA0278P	Spacecraft conical surface pressure 51	B-E-84	N1										X					
CA0279P	Spacecraft conical surface pressure 52	B-E-85	N1										X					
CA0300P	Spacecraft conical surface pressure 53	B-E-86	N1										X					
CA0301P	Spacecraft conical surface pressure 54	B-E-87	N1										X					
CA0302P	Spacecraft conical surface pressure 55	B-E-88	N1										X					
CT0502V	Sync A (90 x 10 commutator) link B	B-E-89	M															
CT0503V	Sync B (90 x 10 commutator) link B	B-E-90	M															



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability																
				1st Order		2nd Order					3rd Order									
				1	2	1	2	3	4	5	6	7	1	2	3	4	5			
TAPE RECORDER TRACK																				
Tape Transport A																				
M																				
CA0120D	X-axis CM display control vib	Track 1	D*															X		
CA0122D	Z-axis CM display control vib	Track 2	D*															X		
CA0124D	CM vib norm to forward sidewall	Track 3	HD*															X		
CA2241D	Unassigned	Track 4																		
	Unassigned	Track 5																		
	Y-axis vib LEB support structure	Track 6	HD*															X		
CK0034Y	Acoustic CM interior	Track 7	D*																	
CT0021V	Link A mixer no. 2 output	Track 8	M*																	
CT0022V	Link B diff pdm (90 x 10) commutator	Track 9	M*																	
CA0136D	CM vib norm to floor	Track 10	HD*																	
CA2240D	Unassigned	Track 11																		
	X-axis vib LEB support structure	Track 12	HD*															X		
CA2242D	Z-axis vib LEB support structure	Track 13	HD*																	
CK0035Y	Acoustic CM interior	Track 14	D*															X		



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order					3rd Order							
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
TAPE RECORDER TRACK/MODULATOR																		
Tape Transport B			M															
SA0940D	SM vibration Sector V (1)	B-1-12	HD*														X	
SA0941D	SM vibration Sector II (2)	B-1-13	HD*														X	
SA0942D	SM vibration Sector II (3)	B-1-14	HD*														X	
SA0924S	Strain SM bend about long axis	B-1-15	HD*														X	
SA0943D	SM vibration Sector II (4)	B-2-12	HD*														X	
SA0944D	SM vibration Sector III (5)	B-2-13	HD*														X	
SA0945D	SM vibration Sector III (6)	B-2-14	HD*														X	
SA0925S	Strain SM bend about cir axis	B-2-15	HD*														X	
SA0946D	SM vibration Sector IV (7)	B-3-12	HD*														X	
SA0947D	SM vibration Sector V (8)	B-3-13	HD*														X	
SA0948D	SM vibration Sector V (9)	B-3-14	HD*														X	
SA0926S	Strain SM long ten/comp	B-3-15	HD*														X	
SA0949D	SM vibration Sector V (10)	B-4-12	HD*														X	
SA0950D	SM vibration Sector V (11)	B-4-13	HD*														X	
SA0951D	SM vibration Sector IV (12)	B-4-14	HD*														X	
SA0927S	Strain SM cir ten/comp	B-4-15	HD*														X	
SA0952D	SM vibration Sector VI (13)	B-5-12	HD*														X	
SA0953D	SM vibration RCS panel (14)	B-5-13	HD*														X	
SA0954D	SM vibration RCS support (15)	B-5-14	HD*														X	



Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order			2nd Order							3rd Order				
				1	2		1	2	3	4	5	6	7	1	2	3	4	5
SA0928S	Strain SM bend about long axis	B-5-15	HD*								X							
SA0934S	Strain SM long ten/comp	B-6-12	HD*								X							
SA0955D	SM vib RCS eng noz X-axis (16)	B-6-13	HD*								X							
SA0956D	SM vib RCS eng noz cir (17)	B-6-14	HD*								X							
SA0929S	Strain SM bend about cir axis	B-6-15	HD*								X							
SA0920S	Strain SM bend about long axis	B-7-12	HD*								X							
SA0921S	Strain SM bend about cir axis	B-7-13	HD*								X							
SA0922S	Strain SM long ten/comp	B-7-14	HD*								X							
SA0930S	Strain SM long ten/comp	B-7-15	HD*								X							
SA0923S	Strain SM cir ten/comp	B-8-12	HD*								X							
SA0932S	Strain SM bend about long axis	B-8-13	HD*								X							
SA0933S	Strain SM bend about cir axis	B-8-14	HD*								X							
SA0931S	Strain SM cir ten comp	B-8-15	HD*								X							
SA0935S	Unassigned	B-9-12	HD*															
SA0936S	Strain SM cir ten/comp	B-9-13	HD*								X							
	Strain SM bend about long axis	B-9-14	HD*								X							
CA2248D	Z-axis CM vibration	B-9-15	HD*								X							X





Table B-2. Operational Readiness of On-Board Instrumentation (Cont)

Measurement No.	Measurement Description	Channel	Operational Readiness	Test Objective Applicability														
				1st Order		2nd Order							3rd Order					
				1	2	1	2	3	4	5	6	7	1	2	3	4	5	
SA0937S	Strain SM bend about cir axis	B-10-12	HD*															
SA0938S	Strain SM long ten/comp	B-10-13	HD*															
SA0939S	Strain SM cir ten/comp	B-10-14	HD*															
SA0958P	SM fluc pressure (2)	B-10-15	HD*															
CT0023V	Link B mixer No. 2 output	B-11	M*															
CT0020V	Link A diff (90x10) commutator	B-12	M*															
CA0121D	Y-axis CM display con vib	B-13	HD*														X	
CA0123D	CM vib normal to heat shield	B-14	HD*											X	X		X	

N1: Thirty-two of 42 conical surface pressure units are mandatory. No two adjacent units can be out.

\*Operational readiness of a tape recorder channel cannot be maintained, but can be determined only by checkout. Operational readiness of the tape transport only can be maintained. Therefore, the readiness of this channel must be established during closeout check of the vehicle in the precount-down task sequence.

N1: Thirty-two of 42 conical surface pressure units are mandatory. No two adjacent units can be out.

\*Operational readiness of a tape recorder channel cannot be maintained, but can be determined only by checkout. Operational readiness of the tape transport only can be maintained. Therefore, the readiness of this channel must be established during closeout check of the vehicle in the precount-down task sequence.



## RANGE INSTRUMENTATION SUPPORT SYSTEMS AND PRIORITY

Electronic Instrumentation

It is mandatory that a sufficient number of WSMR telemetry stations be operational to ensure reception of quality data throughout the flight. A WSMR timing system and liftoff signal are also mandatory.

Photo Sequential and Theodolite Tracking

Table B-3. Operational Readiness of Photographic Tracking

Flight Phase	Vehicle Unit	Cinematic Tracking	Cinetheodolite Positioning
Liftoff	Test vehicle stack	Coverage inclusive from 0 to 200 feet above the pad from 4 stations about the pad M (3) HD (1)	Position recording from 0 to 700 feet above the pad from 4 stations about the pad M (3) HD (1)
Launch (to abort point)	Test vehicle stack	Continuous coverage from approximately 100 feet above the pad up to the abort point (approximately 75,000 ft) from 4 stations M (3) HD (1)	Continuous position recording immediately after liftoff to abort point from 4 stations  M (3) HD (1)
LEV abort and descent	Little Joe II Booster	Continuous coverage of the LJ II from just prior to CM-SM separation to 15 seconds after separation from 2 stations  HD (2)	Continuous position recording from just prior to CM-SM separation to 15 seconds after separation from 3 stations M (2) HD (1)
	LEV	Continuous coverage of the LEV from just prior to CM-SM separation to	Continuous position recording from just prior to CM-SM



Table B-3. Operational Readiness of Photographic Tracking (Cont)

Flight Phase	Vehicle Unit	Cinematic Tracking	Cinetheodolite Positioning
LEV abort and descent (Cont)	LEV (Cont)	LES jettison from 4 stations M (3) HD (1)	separation to LES jettison from 5 stations M (4) HD (1)
LES jettison and CM deceleration	LES	Continuous coverage of the LES from just prior to jettison to 15 seconds after jettison from 2 stations M (1) HD (1)	Continuous position recording from just prior to LES jettison to 15 seconds after jettison from 2 stations HD (3)
	CM	Continuous coverage of the CM from just prior to LES jettison to 15 seconds after CM touchdown from 4 stations M (2) HD (2)	Continuous position recording from just prior to LES jettison to CM touchdown from 4 stations M (3) HD (1)

### Flyby Airplane

Two flyby airplanes are required for preflight calibration of real-time display. One is mandatory, and one is highly desirable.

### FLIGHT DYNAMICS AND WEATHER

#### Launcher Settings

The launcher settings will be 84 degrees in elevation and range center in azimuth (correction will be made for crosswinds to limit crossrange drift to the desired value). The booster control system ensures a downrange abort point including booster and wind dispersions.

Note that the Little Joe II booster contains an attitude control system and does not require changes in launch angle as a function of winds. Only range, crossrange, and flight path angle are affected.



### Abort Conditions\*

A 15-degree pitchup maneuver by the Little Joe II booster just prior to launch escape vehicle abort is necessary to ensure vehicle tumble at the tumbling boundary altitude for development of high-pressure loads on the escape vehicle. The pitchup will be performed by the booster upon receipt of radio command signal as the booster attains an altitude of 71,000 feet.

Actual abort will be initiated 3.5 seconds after the pitchup maneuver is begun by the booster-contained abort delay timer activated simultaneously with receipt of the pitchup signal.

A backup timer in the Little Joe II will furnish the signal for pitchup and activation of the abort delay timer upon failure of the radio command to accomplish this function.

### Wind Restrictions

The allowable surface wind velocity is 23.7 knots (27.3 mph) maximum to control horizontal touchdown velocity and eliminate towing of the command module by the parachutes.

During the 40 minutes prior to launch, any unstable or gusty wind condition that exceeds the allowable wind limits will be cause for a hold until the weather becomes stable. Allowable winds aloft must not be greater than WSMR 95 percent Cumulative Percent Frequency (CPF) wind velocity of worst month of year combined with WSMR 99 percent CPF wind shear.

### Photographic Requirements

Uninterrupted photographic tracking is required to 120,000 feet above the launch pad and up to 40 nautical miles in range. Any weather conditions that would inhibit these requirements will be cause to hold or scrub the test.

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\*Stated conditions may vary slightly pending receipt of NASA Little Joe II boost trajectories.

## APPENDIX C

### WSMR APOLLO FLIGHT PROGRAM



NORTH AMERICAN AVIATION, INC.

SPACE and INFORMATION SYSTEMS DIVISION

Table C-1. WSMR Apollo Flight Program Mission Objectives

DEVELOPMENT ISSUES	PAD ABORT PA-1 BOILERPLATE 6	TRANSONIC ABORT A-001 BOILERPLATE 12	HIGH q ABORT 002 BOILERPLATE 23	HIGH ALTITUDE ABORT A-003 BOILERPLATE 22	POWER-ON TUMBLING BOUNDARY ABORT A-004 S/C 002 OR A-004A S/C 010	PAD ABORT PA-2 BOILERPLATE 23A
OBJECTIVES						
CAPABILITY (TO ABORT) LES	<ul style="list-style-type: none"><li>● Demonstrate capability of the escape system to abort the Apollo escape configuration during a pad abort</li><li>● Demonstrate proper operation of lower jettison and pitch control action</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the capability of the escape system to abort the CM entry away from the launch vehicle</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory LES performance in the maximum dynamic pressure region with conditions approximating EDS limits</li><li>● Demonstrate the performance of the LES in the maximum dynamic pressure region</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory LES performance in the maximum dynamic pressure region with conditions approximating EDS limits</li><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory LES performance in the maximum dynamic pressure region with conditions approximating EDS limits</li><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate capability of the LES to abort the launch pad and recover</li></ul>
STABILITY LEV	<ul style="list-style-type: none"><li>● Determine aerodynamic stability characteristics of the Apollo escape configuration during a pad abort</li></ul>	<ul style="list-style-type: none"><li>● Determine aerodynamic stability characteristics of the Apollo escape configuration for this abort condition</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory LES performance in the maximum dynamic pressure region with conditions approximating EDS limits</li><li>● Demonstrate the performance of the LES in the maximum dynamic pressure region</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li><li>● Demonstrate the capability of the LES to a main engine shutdown and recovery</li></ul>	<ul style="list-style-type: none"><li>● Determine performance and stability characteristics of the LES with respect to abort from the launch pad and recover</li></ul>
INTEGRITY (STRUCTURAL PERFORMANCE)		<ul style="list-style-type: none"><li>● Demonstrate the structural integrity of the escape tower</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the structural performance of the LES with the sound subsystem</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the structural integrity of the LES during launch and entry from high altitude</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the structural integrity of the LES during launch and entry from high altitude</li></ul>	<ul style="list-style-type: none"><li>● Determine performance and stability characteristics of the LES with respect to abort from the launch pad and recover</li></ul>
SEPARATION	<ul style="list-style-type: none"><li>● Demonstrate proper operation of the launch escape tower release device</li><li>● Determine initial separation trajectory of the launch escape tower</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory recovery timing sequence in the LES</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory separation of the LES plus boat protective cover from the CM</li><li>● Demonstrate satisfactory separation of the LES from the SM at an angle of attack</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the LES plus boat protective cover after high altitude entry</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the LES plus boat protective cover from the CM</li><li>● Demonstrate satisfactory separation of the LES</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory separation of the LES plus boat protective cover from the CM</li><li>● Demonstrate satisfactory separation of the LES</li></ul>
RECOVERY ELS	<ul style="list-style-type: none"><li>● Demonstrate earth landing timing sequence and proper operation of the parachute subsystem of the ELS</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate proper operation of the CM-SM separation subsystem</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the LES using retro dual drogues</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate the LES using retro dual drogues</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate satisfactory operation and performance of the ELS with a spacecraft vehicle</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate proper earth landing sequence during launch escape and recovery</li></ul>
SEQUENCE (OF EVENTS)	<ul style="list-style-type: none"><li>● Demonstrate launch escape timing sequence</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate proper operation of the applicable components of the ELS</li></ul>				
ENVIRONMENT (EFFECTS INDUCED)	<ul style="list-style-type: none"><li>● Determine dynamics of the CM during jettisoning of escape tower</li><li>● Determine escape tower vibration during a pad abort</li></ul>	<ul style="list-style-type: none"><li>● Determine aerodynamic loads due to local flow on the CM and SM during a Little Joe II launch</li></ul>	<ul style="list-style-type: none"><li>● Determine the CM pressure loads including dynamic pressure region, in the maximum dynamic pressure region</li><li>● Determine the aerodynamic pressure loads on the CM during the launch phase</li><li>● Obtain thermal effect data on the CM during on abort in the maximum dynamic pressure region</li></ul>	<ul style="list-style-type: none"><li>● Obtain data on thermal effect during launch and during reentry of the launch escape tower</li><li>● Determine the dynamic loads and the structural response of the CM during launch and high altitude abort</li><li>● Determine vibration and acoustic environment and response of the SM with simulated RCS quads</li></ul>	<ul style="list-style-type: none"><li>● Determine the static loads on the CM during launch and the abort sequence</li><li>● Determine the dynamic loads on the CM during launch and the abort sequence</li><li>● Determine the static loads and the structural response of the CM during launch and the abort sequence</li><li>● Obtain data on the structural response of the CM during a power-on tumbling abort</li><li>● Obtain thermal data on the BPC during a power-on tumbling abort</li><li>● Obtain structural noise data inside the CM at an astronaut station</li></ul>	<ul style="list-style-type: none"><li>● Determine the static loads on the CM during launch and the abort sequence</li><li>● Determine the dynamic loads on the CM during launch and the abort sequence</li><li>● Determine the static loads and the structural response of the CM during launch and the abort sequence</li><li>● Obtain data on the structural response of the CM during a power-on tumbling abort</li><li>● Obtain thermal data on the BPC during a power-on tumbling abort</li><li>● Obtain structural noise data inside the CM at an astronaut station</li></ul>
COMPATIBILITY BOOSTER, GSE, ETC.	<ul style="list-style-type: none"><li>● Demonstrate operation of the research and development instrumentation and communications equipment to be used on subsequent flights</li><li>● Demonstrate compatibility of prototype handling GSE</li></ul>	<ul style="list-style-type: none"><li>● Demonstrate Little Joe II-S/C compatibility</li></ul>				
MISCELLANEOUS						
STATUS	NASA PROJECT APOLLO WORKING PAPER NO. 1040C, 20 OCTOBER 1963 MISSION ACCOMPLISHED NOVEMBER 1963	WP 1005A, 15 APRIL 1964, WITH REVISION PER NASA TWR 1009, 7 MAY 1964 MISSION ACCOMPLISHED 12 MAY 1964	NASA TWR 1033-1032-04-1038 (W7780MA), 20 AUGUST 1964	NASA TWR 1033-1032-04-1032 (W7802MA), 20 AUGUST 1964	NASA TWR 1033-1032-04-1032 (W7802MA), 20 AUGUST 1964	NASA TWR 1033-1032-04-1032 (W7802MA), 20 AUGUST 1964
● FIRST-ORDER TEST OBJECTIVE      ○ SECOND-ORDER TEST OBJECTIVE      ○ THIRD-ORDER TEST OBJECTIVE						
PREPARED BY: DEVELOPMENT ANALYST DFT 972-594, EXT 3422, 3423						
COMPLETION DATE: 18 JUNE 1964 REVISION DATE: 1 FEBRUARY 1965						

C-1, C-2

SID 64-2174



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SPACE and INFORMATION SYSTEMS DIVISION

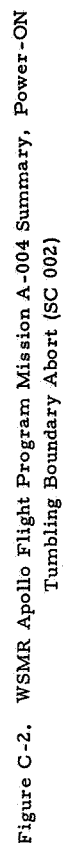
	MISSION PA-1 (PAD ABORT) BOILERPLATE 6	MISSION A-001 (TRONSONIC ABORT) BOILERPLATE 12	MISSION A-002 (HIGH Q ABORT) BOILERPLATE 23	MISSION A-003 (HIGH ALTITUDE ABORT) BOILERPLATE 22	MISSION A-004 POWER-ON TURNING BOUNDARY ABORT SPACECRAFT 002	MISSION PA-2 (PAD ABORT) BOILERPLATE 23A
LAUNCH ESCAPE SYSTEM	TOWER ATTACH: SINGLE MOTOR BOLTS ATTENTION MOTOR: THIRST OFFSET ANGLE 2.3°	TOWER ATTACH: SINGLE MOTOR BOLTS ATTENTION MOTOR: THIRST OFFSET ANGLE 1.5°	TOWER ATTACH: SINGLE MOTOR BOLTS ATTENTION MOTOR: THIRST OFFSET ANGLE 4° CANARDS	A 319 A 319 A 300 CANARDS	A 319 A 319 A 300 CANARDS	TOWER ATTACH: SINGLE MOTOR BOLTS ATTENTION MOTOR: THIRST OFFSET ANGLE 4° CANARDS
EARTH LANDING SYSTEM	DOWN CHUTE: SINGLE 13.7" DIA., HOT REED PITCH CHUTES: 15° DIA., 15° HORISZ., ALL VERTICALLY MAIN CHUTES: SINGLE 13.7" DIA., 1 LEGS/MINUTE, 1000 ANGLES 5°	SINGLE CHUTE: SINGLE 13.7" DIA., HOT REED PITCH CHUTES: 15° DIA., 15° HORISZ., ALL VERTICALLY MAIN CHUTES: SINGLE 13.7" DIA., 1 LEGS/MINUTE, 1000 ANGLES 5°	DOWN CHUTE: SINGLE 13.7" DIA., HOT REED PITCH CHUTES: 15° DIA., 15° HORISZ., ALL VERTICALLY MAIN CHUTES: SINGLE 13.7" DIA., 1 LEGS/MINUTE, 1000 ANGLES 5°	DOWN CHUTE: SINGLE 13.7" DIA., HOT REED PITCH CHUTES: 15° DIA., 15° HORISZ., ALL VERTICALLY MAIN CHUTES: SINGLE 13.7" DIA., 1 LEGS/MINUTE, 1000 ANGLES 5°	DOWN CHUTE: SINGLE 13.7" DIA., HOT REED PITCH CHUTES: 15° DIA., 15° HORISZ., ALL VERTICALLY MAIN CHUTES: SINGLE 13.7" DIA., 1 LEGS/MINUTE, 1000 ANGLES 5°	DOWN CHUTE: SINGLE 13.7" DIA., HOT REED PITCH CHUTES: 15° DIA., 15° HORISZ., ALL VERTICALLY MAIN CHUTES: SINGLE 13.7" DIA., 1 LEGS/MINUTE, 1000 ANGLES 5°
COMMAND MODULE	HEAT SHIELD: SPARKS WITH TOWER ECL. COMPARE WITH WATER CIRCULATION RECORDING, 1-10 10 RECORD, 1-10 10 EPI. BATTERIES, 2 PROTONIC 1-10 10 1-10 10	HEAT SHIELD: SPARKS WITH TOWER ECL. COMPARE WITH WATER CIRCULATION RECORDING, 1-10 10 RECORD, 1-10 10 EPI. BATTERIES, 2 PROTONIC 1-10 10 1-10 10	HEAT SHIELD: SPARKS WITH TOWER ECL. COMPARE WITH WATER CIRCULATION RECORDING, 1-10 10 RECORD, 1-10 10 EPI. BATTERIES, 2 PROTONIC 1-10 10 1-10 10	HEAT SHIELD: SPARKS WITH TOWER ECL. COMPARE WITH WATER CIRCULATION RECORDING, 1-10 10 RECORD, 1-10 10 EPI. BATTERIES, 2 PROTONIC 1-10 10 1-10 10	HEAT SHIELD: SPARKS WITH TOWER ECL. COMPARE WITH WATER CIRCULATION RECORDING, 1-10 10 RECORD, 1-10 10 EPI. BATTERIES, 2 PROTONIC 1-10 10 1-10 10	HEAT SHIELD: SPARKS WITH TOWER ECL. COMPARE WITH WATER CIRCULATION RECORDING, 1-10 10 RECORD, 1-10 10 EPI. BATTERIES, 2 PROTONIC 1-10 10 1-10 10
SERVICE MODULE	MAIN: MASSIVE TRACKING NOT REQUIRED	MAIN: TWO C-BAND TRANSFORMERS REINFORCED BLAST BARRIAD INSULATED 40' CUBES	MAIN: TWO C-BAND TRANSFORMERS REINFORCED BLAST BARRIAD INSULATED 40' CUBES	MAIN: TWO C-BAND TRANSFORMERS REINFORCED BLAST BARRIAD INSULATED 40' CUBES	MAIN: TWO C-BAND TRANSFORMERS REINFORCED BLAST BARRIAD INSULATED 40' CUBES	MAIN: TWO C-BAND TRANSFORMERS REINFORCED BLAST BARRIAD INSULATED 40' CUBES
BOOSTER OR INTERFACE	NO BOOSTER REQUIRED ON TO PAD ABORT USED.	BOOSTER: LITTLE JOE II MOTOR: 1 ALCOOL, 4 RECUIT THIRST TEMPERATURE: 1000, 1000 AT CONTROL: 4 FMS WITH VARIABLE POL. CONTROL: 4 FMS WITH VARIABLE POL. INSTR. ON BOARD 7M	BOOSTER: LITTLE JOE II MOTOR: 1 ALCOOL, 4 RECUIT THIRST TEMPERATURE: 1000, 1000 AT CONTROL: 4 FMS WITH VARIABLE POL. CONTROL: 4 FMS WITH VARIABLE POL. INSTR. ON BOARD 7M	BOOSTER: LITTLE JOE II MOTOR: 1 ALCOOL, 4 RECUIT THIRST TEMPERATURE: 1000, 1000 AT CONTROL: 4 FMS WITH VARIABLE POL. CONTROL: 4 FMS WITH VARIABLE POL. INSTR. ON BOARD 7M	BOOSTER: LITTLE JOE II MOTOR: 1 ALCOOL, 4 RECUIT THIRST TEMPERATURE: 1000, 1000 AT CONTROL: 4 FMS WITH VARIABLE POL. CONTROL: 4 FMS WITH VARIABLE POL. INSTR. ON BOARD 7M	NO BOOSTER REQUIRED ON TO PAD ABORT USED
COMPLETION DATE: DEC 15, 1964						

NOTE:  
THIS DRAWING IS SUBJECT TO  
THE CONFIDENTIALITY OF THE  
LITTLE JOE II

Figure C-1. WSMR Apollo Flight Program Vehicle Delta Configuration

C-3, C-4

SID 64-2174



C-5, C-6

SID 64-2174